A Report

FEASIBILITY STUDY OF AN AERONOMY SATELLITE

FINAL REPORT

TO

THE NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

GODDARD SPACE FLIGHT CENTER

UNDER CONTRACT NO. NAS5-9346

5 AUGUST 1966

Space Craft, Inc.

INTRODUCTION

This report presents the findings of a preliminary system definition study, conducted to support a Goddard Space Flight Center objective in aeronomy. The aeronomy objective is to investigate the properties of that region of the atmosphere whose lower limit overlaps regions accessible by sounding rockets and whose upper limit extends into regions presently mapped by aeronomy satellites.

The study examines two satellite concepts in which the energy lost to drag is periodically restored by means of a multiple-start propulsion system. Orbit lifetimes are considerably increased over the non-propelled case for a given mission profile. The two satellite concepts, the study guidelines, and the tasks accomplished are identified below.

The Aeronomy/550 Satellite

The first Satellite concept which was investigated had been proposed to Goddard Space Flight Center by Space Craft, Inc. in February 1965. The present scope of work required an adaption of this concept to Scout or Thor-Delta launch vehicles. The adaptions were executed and are designated Aeronomy/550S and Aeronomy/550 T respectively. Design is characterized by articulated Solar Panels and a local vertical stabilization scheme using active horizon sensors as well as gas jets operating in a limit cycle.

The Aeronomy/SS Satellite

The second satellite concept which was investigated is based on further guidelines developed in a meeting between personnel of Goddard Space Flight Center and of Space Craft, Inc. The guidelines are directed at greater mechanical simplicity and reliability, and call for a spin stabilized satellite

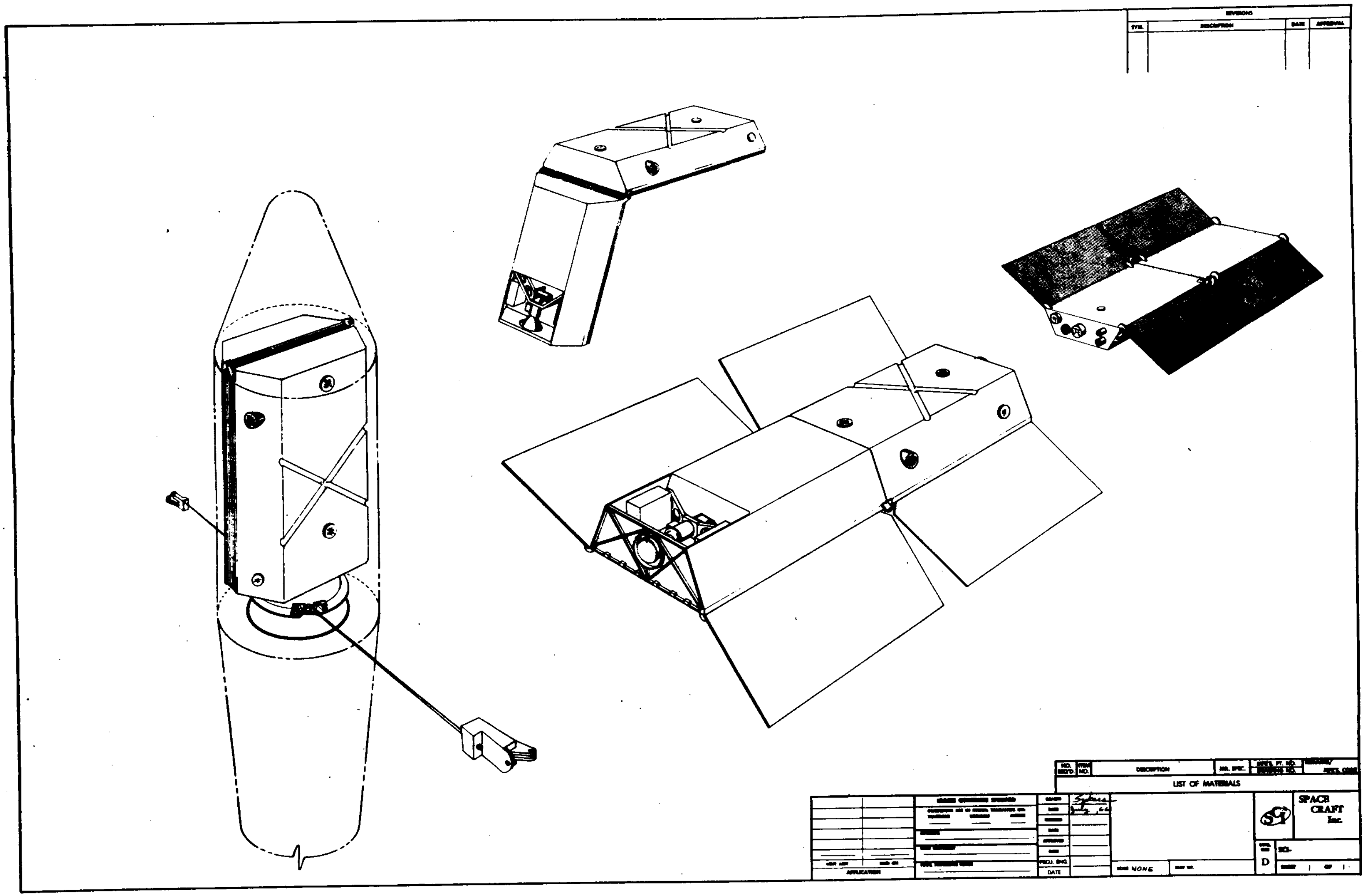


FIGURE 1-2
ARTIST CONCEPTION AERONOMY/550 SATELLITE

to be adapted to Thor-Delta launch vehicles. A 700 lbm. design is investigated for launch on a long-tank vehicle, and a 1200 lbm. design for launch on the same vehicle with thrust-augmenting solid propellant boosters. The Aeronomy/SS designs are referred to as a "Target" design if conservative structural mass estimates are used; and as " optimum" design if minimum mass estimates are used. The spin axis is maintained normal to the orbit plane by an ACS which uses passive aspect sensors and a two-pulse, two-jet RCS. Solar cells are fixed. The configuration is shown in Figure 1-1.

Technical Guidelines

The guidelines for the Aeronomy/SS approach include the following, in addition to spin stabilization:

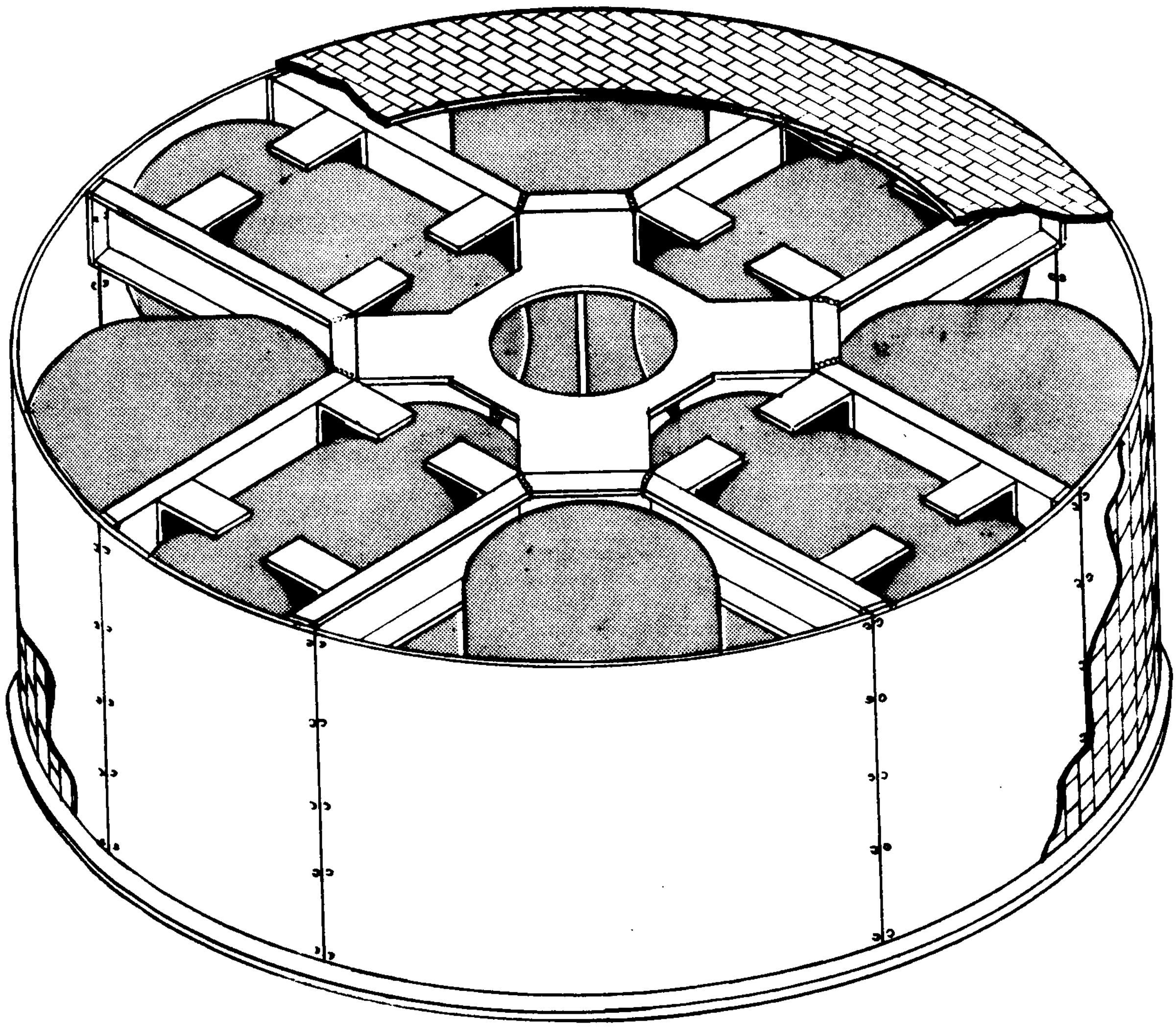
Satellite useful Lifetime	1 year or more approx. 250 km Perigee to 800 km Apogee
Mission Orbit Profile	Approx. 120 km Perigee 10 to 50 orbits per Perigee Lowering
Mission Maneuver Profile	
Mission Maneuver Frequency	Approx. 25 maneuvers per year
Experiment Mass	Up to 50 lbs.
Experiment Power	5 to 10 watts, average up to 100 + watts peak
Experiment Volume	2 ft.^{3} nominal
Experiment "ON" time	2 hours/day, minimum
Launch Vehicle Selection	1. Improved Delta, Minimum Staging preferred 2. Scout 3. Saturn, Bonus payload.

The preliminary systems definition tasks required to be documented are:

1. Define program objectives and guidelines
2. Perform mission analysis
3. Establish design and operational criteria

4. Prepare systems specification
5. Furnish preliminary design aeronomy/550 SPM
6. Define and schedule major program
7. Develop program plan
8. Develop cost estimates

The results of the study are presented in essentially this order in the body of the report. It is found that the propelled aeronomy satellite concept for low atmosphere investigations is entirely feasible.



AERONOMY/SS, STRUCTUAL CONCEPT

SUMMARY

This brief survey of the potential application of propulsion to aeronomy satellites finds that the concept is entirely feasible and merits detailed study. The inclusion of a propulsion system provides for very low perigee orbits that extend data gathering excursions well under present altitude limits. While it is not the purpose of this study to provide detailed design data, sufficient evidence is presented to support the conclusions.

2.1 PRINCIPAL PARAMETER INFLUENCES ON MISSION

Sufficient orbital and configuration studies are included to permit certain general conclusions. A Basic Mission profile is selected as a yardstick in these parametric analyses. This Basic Mission consists of performing 15 low-perigee sweeps once every 15 days for one year; the remaining days of the year are spent in a higher-perigee parking orbit. Injection is assumed to establish the parking orbit with a 250 km perigee and an 800 km apogee. The conclusions are:

a) for the Basic Mission only about 15 percent of the propellant is used directly in compensating for drag. The remaining propellant supports orbit changes to avoid drag.

b) for a mission with a parking perigee of 200 km, about 30 percent of the propellant is used directly in compensating for drag, but mission life is extended about 30 percent despite this.

c) lowering apogee below about 800 km is not advisable inasmuch as late-life decay during sweeps becomes critical.

d) raising injection apogee is always beneficial - it increases satellite initial energy and staves off final decay.

e) raising propellant mass fraction is always beneficial, and for the same reason.

f) configuration constraints are only mild within reasonable limits.

It is concluded that a time-effective mission profile will emphasize lowest practical parking perigee; and highest injection apogee; together with the best compromise of high propellant mass fraction, packaging convenience, and lastly minimum drag area. The question of whether the most time-effective orbit is also the best data-gathering orbit must be left for further definition by the experimenter.

2.2 AERONOMY/SS CONCEPT AND CAPABILITIES

Figure 1-1 depicts the spin-stabilized configuration having a 1200 lbm initial mass. The cylindrical, disc-shaped satellite displays solar cells on the entire circumference and an equal area, around the "upper" and "lower" rim sections. Experiment instruments are principally mounted on the equator. Satellite systems consist of data and communications subsystems; an ACS System with its electronics, IR horizon sensor and sun sensor for attitude sensing, and two reaction control thrusters which use the main engine propellants for spin axis orientation; and a propulsion system. The latter consist of an engine and liquid propellant tanks closely nested in the major structure of the satellite. A Yo-Yo despin is provided to remove the initial angular momentum. Fine despin is accomplished with small rim-mounted jets. Magnetic RCS is not considered due to the high magnet mass necessary to meet time-to-turn requirements.

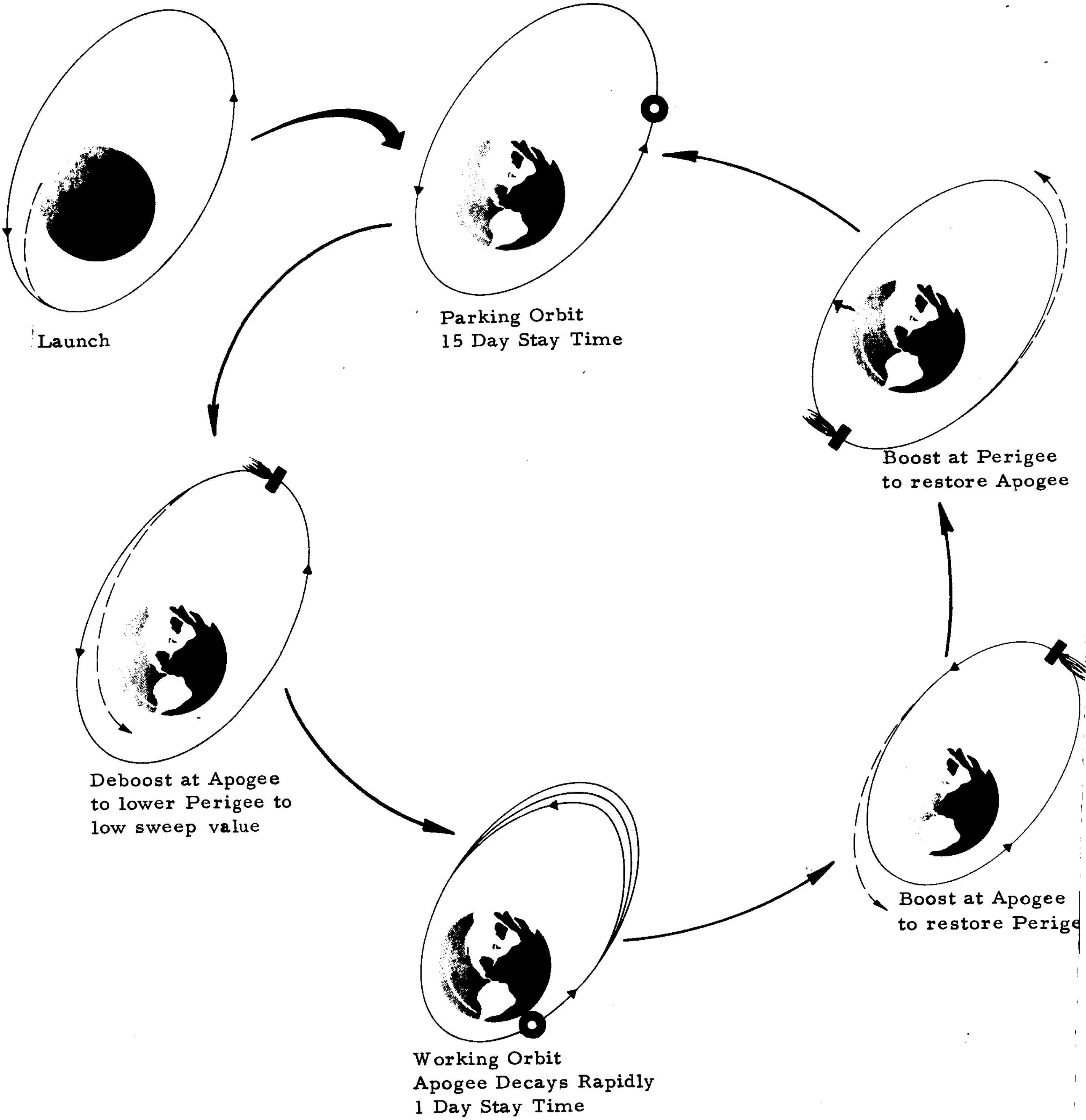
The propulsion system thrust axis points along the spin axis, but the spin axis is normal to the desired line of thrust during most of the satellite's life. When a propulsion maneuver is commanded, the thrust-spin axis is precessed to the desired orientation by a two-pulse ACS schedule executed by the two ACS thrusters. This schedule must be completed in considerably less time than half an orbit if maximum satellite life is desired. A full maneuver to lower perigee, raise same, and restore apogee involves three propelled phases, each of which requires axis orientation before and after propulsion. The maneuver sequence is indicated in figure 2-1.

Three specific design points were investigated for the Aeronomy/SS: a 700 lbm design, a 1200 lbm "Target" design, and a 1200 lbm "Optimum" design. For each of these both hydrazine monopropellant and a bipropellant were investigated. The essential differences between target and optimum designs are the structural and tank masses allowed. The target design should be readily producible; the optimum designs may require rigorous weight control programs, plus tank development beyond that necessary for the target designs.

Further systems details are summarized in Section 3. 0.

2.3 AERONOMY/550 CONCEPT AND CAPABILITIES

Figure 1-2 depicts the three-axis stabilized 550 approach adapted to the Scout vehicle. The configuration consists of one forward and one aft satellite body section, each bearing two solar panels hinged to their sides. The forward section contains all spacecraft electronics and the aeronomy experiments. The aft section contains propulsion system, RCS systems, and part of the battery mass necessary for balancing. The satellite body sections are folded over the hinged solar panels. After payload separation, the



MANEUVER SEQUENCE

FIGURE 2-1

Yo-Yo despin system is released and the anguler momentum is removed from the payload. Fine despin is provided by the RCS. The satellite is then unfolded and oriented to initiated the mission.

For Thor-Delta launch, the solar panels are formed as single units per side, and the body is not hinged. This represents considerable simplification. The body is extended to 100 inches to permit storing of more propellant. The resulting configuration has a lifetime meeting the Basic Mission requirement, and is comparable to the optimum 1200 Aeronomy/SS design.

Since the thrust axis need not be rotated, certain maneuver simplifications result. The principal benefit is probably felt by the ground control net: absence of the axis-rotation requirement relieves both timing and data position determination requirements somewhat. This possible advantage is balanced by greater mechanical complexity in the solar panels, and the ACS.

Further systems details are summarized in Section 3.

2.4 TYPICAL MISSIONS AND OPERATIONS SEQUENCE

The sequence executed by a propelled aeronomy satellite consists of the following:

0. (Launch to) elliptic parking orbit
1. Retrograde apogee kick to lower perigee
2. Working mission as apogee decays
3. Kick in apogee to restore perigee
4. Kick in perigee to restore apogee.

Steps 1 through 4 constitute the altitude-changing mission maneuver executed each time a low-perigee sweep sequence is performed, and is common to the design concepts presented in this report. This sequence is illustrated for the Aeronomy/SS concept in figure 2-1. Note that both before and after each propulsion step (1, 3, 4 in above list) a 90^{o} thrust axis turn in attitude must be made, for a total of 540^{o} per 'maneuver'. For the local-vertical stabilized 550 version, a 180^{o} turn must be made before and after step 1.

The integration of this mission sequence into the launch vehicle and ground station control is shown in figure 2-2. Specific operational support required from the ground station is further broken out in figure 2-3. Further details are found in paragraph 3.1.1 of the specifications.

It is judged that the overall impact on the operations net will be felt more in the frequency of maneuver control demand, than in the nature of the demand. The relatively short time available to determine and command to the satellite the required parameters also merits more detailed study.

2.5 PROGRAM PLAN AND COST ESTIMATES

Figure 2-4 summarizes the major elements of a 3 1/2 year program plan to orbit a propelled aeronomy satellite by 1970. It is estimated that tailoring the propulsion and RCS systems will be the pacing item for the development phase; and that otherwise the schedule provides relatively generous times between milestones. Some compression of the schedule therefore would be feasible. Further details are found in Section 7 of this report.

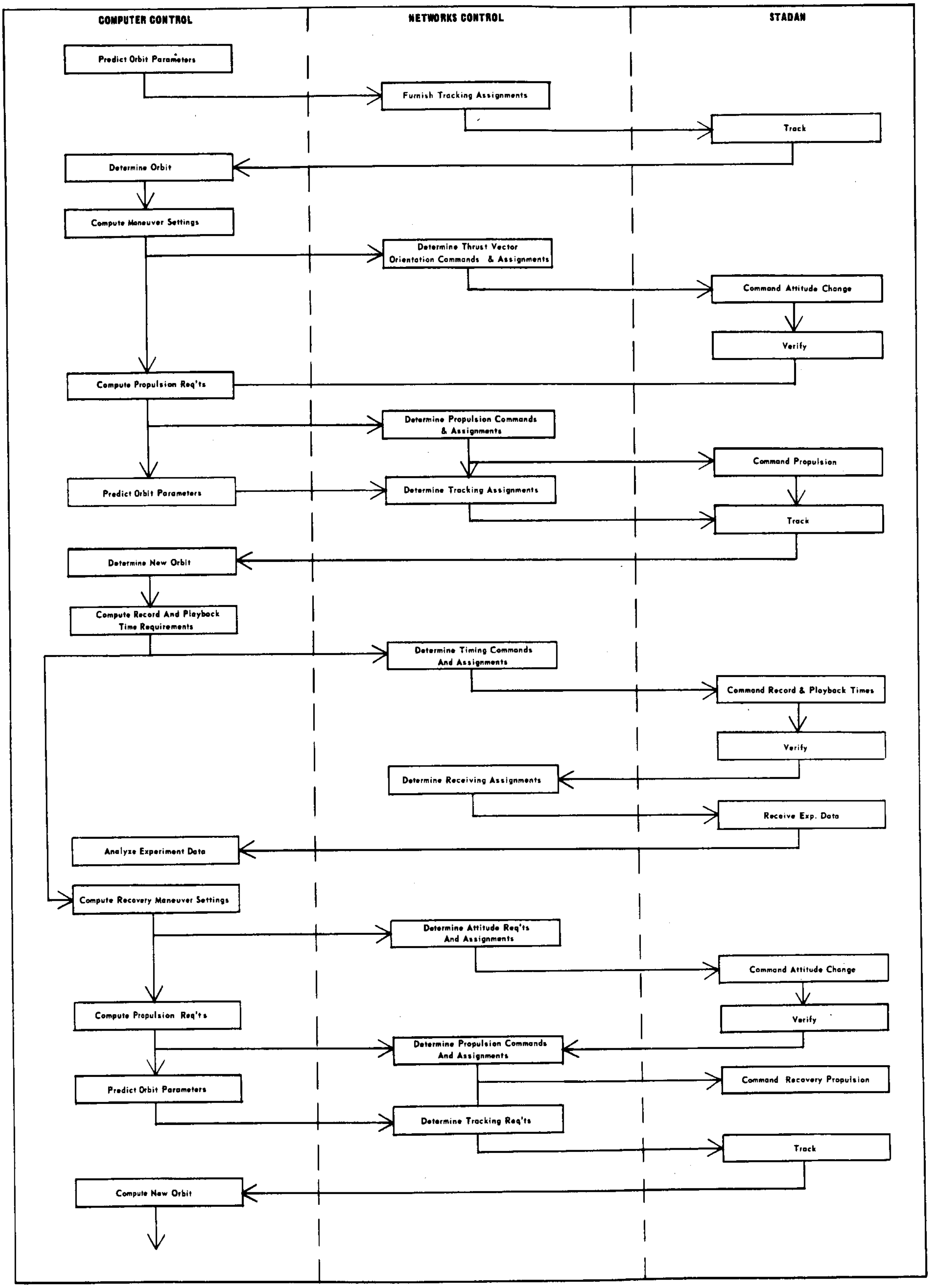
The schedule begins with a four-month final definition study. Timely accomplishment of the total program for a 1969 launch rests strongly on completion of the decisive definition phase, and early initiation of this program element is recommended.

MISSION OPERATIONS PROFILE

LAUNCH VEHICLE	SATELLITE					GROUND STATION			
	Propulsion & A.C.S.	PM TRANSMISSION	FM TRANSMISSION	COMMAND RECEPTION	DATA	TRACK	PM RECPT.	FM RECEPT.	COMMAND
		Transmit Real Time				X	X		
Provide Parking Orbit									
Establish Initial Attitude									
Eject Shroud									
Eject Satellite									
	Maintain Attitude								
				Receive Propulsion Settings					X
	Orient Thrust Axis								
				Receive Propulsion Execute					X
	Lower Perigee								
	Reestablish Attitude								
				Receive Timing Command					X
			Transmit Stored Data		Record Data			X	
			⋮		Reproduce Data			⋮	
				Receive Orientation Settings	⋮				X
				Receive Propulsion Sequence					X
	Orient Thrust Axis								
	Raise Perigee								
				Receive Orientation Settings					X
				Receive Propulsion Sequence					X
	Raise Apogee								
	Reestablish Attitude								

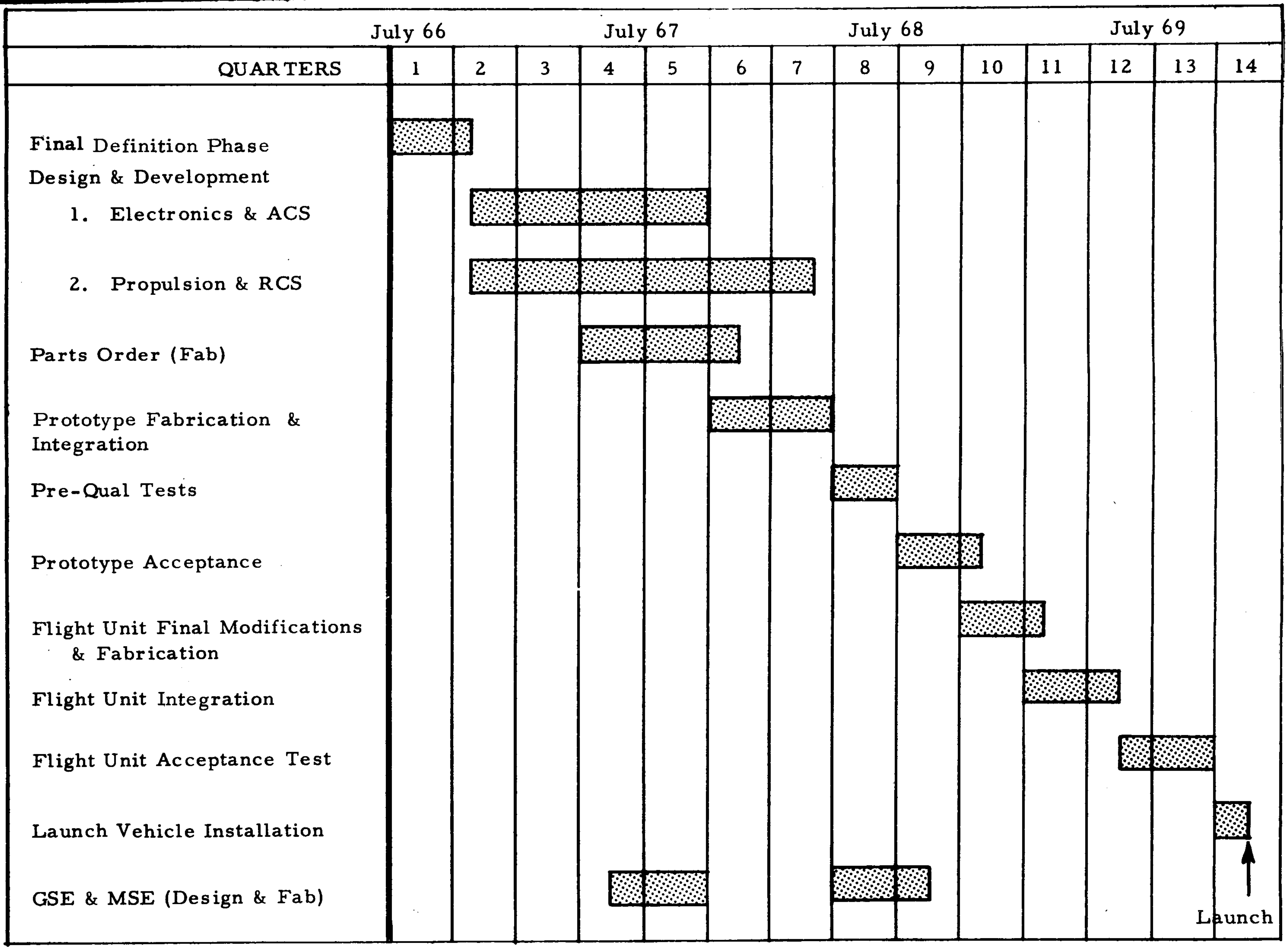
MISSION OPERATIONS PROFILE

FIGURE 2-2



MISSION FLIGHT OPERATIONS AND SUPPORT SEQUENCE

FIGURE 2-3



MAJOR MILESTONE CHART
FIGURE 2-4

The brief preliminary cost analysis presented in Section 7 of this report indicates that the development program through prototype delivery, including one set of support equipment, would cost approximately 2.7 million dollars. A continuation of the program as indicated on the milestone chart of figure 2-4 adds the flight satellite plus an additional support equipment set for about 1.8 million dollars. The contract-basis cost estimate is therefore approximately 4.5 million dollars. These estimates will be revised downward if strong government participation is elected.

2.6 RECOMMENDATIONS

Technical and cost effectiveness considerations indicate that the Aeronomy/SS and the Aeronomy/550T 1200 lbm designs, both of which can meet the defined basic mission, should be prime goals for further study.

The major differences in these designs are in the respective ACS-RCS and solar panel arrays: a cost and reliability comparison of these subsystems is recommended, to include ground net loops.

Since the 700 lbm Aeronomy/SS design comes close to meeting the defined basic mission, a quick look to evaluate further orbit parameter changes is indicated. This is the lowest-cost system.

Design-peculiar mission profile advantages should be identified for each design. This plus the preceding task then selects a single design approach.

Detailed mission profiles for the single approach are carried out, to achieve a best-profile mission using updated atmosphere. Toward the end of this phase, a selection of monopropellant or bipropellant system is made.

Detailed design approach is confirmed, then detailed to the extent necessary to permit concept approval. Emphasis on tank and structure mass reduction is recommended.

Systems Engineering effort is recommended to establish and confirm realistic schedules, cost, and documentation requirements. A minimal PERT-TIME and PERT-COST or equivalent system should be outlined.

-3-

SYSTEM CRITERIA

The System Criteria for the Aeronomy Satellite are summarized in Table 3-1. The five configurations studied are presented in tabular form for ease of comparison. The basic configurations are based on the maximum launch capability of several standard launch vehicles.

Mass Breakdown Tables are given in Tables 3-2 through 3-6 for each configuration with each subsystem of that configuration reduced to its basic constituents.

The power requirements for each of the two systems (local vertical and spin stabilized) are given in Table 3-7 and 3-8.

The propellant capacity in mass units is specified as a parameter of launch weight in figure 3-1 for Aeronomy/SS hydrazine systems, and in figure 3-2 for bipropellant systems. Corresponding relations for volume of propellant are given in figures 3-3 and 3-4. These figures are used to enter the mission lifetime tables of Section 4, and to correlate to the dimensional tables of Appendix B. The governing equations are given in Section 5.2.1 together with the propulsion and structures systems constants used to evaluate them.

Further system criteria are presented in the Aeronomy Satellite Specifications, Part I in Section 6.0 of this report.

Launch Vehicle		Scout	2 Stage Thor Delta	Improved TAD		
Characteristics (1)		Aeronomy/550 S	"700" Aeronomy	"1200" Target	"1200" Optimum	Aeronomy/550 T
Satellite Gross Weight (2)		380 lbm	700 lbm	1200 lbm	1200 lbm	1200 lbm
% Basic Mission Life (3)	Hydrazine (4)	N/A	56	68	80	N/A
	Bipropellant (5)	16	72	88	108	105
Experiment Weight		50 lbm	40 lbm	50 lbm	40 lbm	50 lbm
Experiment Volume		1.7 cu. ft.	1.3 cu. ft.	2.6 cu. ft.	2.7 cu. ft.	1.7 cu. ft.
Overall Dimensions L x W x H (in)		Body 40 x 28 x 10.1 Deployed 40x60x10.1	44.6 (D) x 12.1	60 (D) x 13.4	60 (D) x 15.9	100 x 60 x 10.1
Maneuver Capability (ΔV Nominal Total)		1 50 ft/sec.	7500 ft/sec.			7,500 ft/sec.
Attitude Control & Stabilization System Cruise Mode		Local Vertical Autopilot Reference to local vertical $\pm 2^{o}$ to orbital velocity vector $\pm 2^{o}$ Autopilot capability; command positioning and holding $\pm 3^{o}$	Spin Angle between spin axis and normal to orbit plane 2^{o}			Same as Aeronomy/550S on Scout
Power System	Average Power to Exp.	44 watts	5 watts	10 watts	10 watts	38 watts
	Solar Array Area	37 ft^2 (total)	3 ft^2 (projected)	4.6 ft^2 (projected)	4.6 ft^2 (projected)	37 ft^2 (total)
Data System	Data Rate	2800 bits/sec.	8640 bits/sec	8640 bits/sec	8640 bits/sec.	2800 bit/sec.
	Stored Data	Orbit Coverage	25 min.	25 min.	25 min.	Orbit Coverage

(1) All Command Capability STADAN Compatible
(2) Excludes Adapter and Despin Mechanism
(3) Basic Mission: Parking Orbit 250 KM Perigee 800 KM Apogee; Low Sweep Perigee 120 KM; Number of Maneuvers, 25; Number of Sweeps per Maneuver, 15; Number of days between Maneuvers, 15.
(4) I_{SP} = 225
(5) I_{SP} = 300

AERONOMY SATELLITE FACT SHEET

TABLE 3-1

	lbm	kg
Data System	(10.6)	(4.81)
Tape Recorder	6.6	
Signal Conditioner	1.0	
Encoder	3.0	
Communications System	(7.50)	(3.4)
Transmitter Hi Power	.75	
Transmitter Lo Power	.50	
Command Receiver	1.5	
Command Decoder	3.0	
Antennas	1.0	
Diplexer	.25	
Hybrid Ring	.5	
Power System	(82.30)	(37.36)
System Batteries	50.0	
Ordnance Batteries	2.0	
Solar Cell Array	23.3	
Battery Charger	.5	
Regulator	.5	
DC/DC Converter	.5	
DC/AC Converter	.5	
Power Distributor	5.0	
Propulsion System	(43.3)	(19.98)
Engine	6.3	
N_2 Tanks	19.0	
N_2	3.0	
Valves, Filters, Plumbing	15.0	
Attitude Control System	(20.0)	(9.08)
ACS Electronics	2.5	
Horizon Sensors (3)	10.5	
Yaw Gyro Compass	2.0	
Autopilot		
Rate Gyros	1.5	
Integrating Accelerometer	1.0	
Internal Sensor Housing	2.5	
Reaction Control System	(34.6)	(15.71)
Tanks	20.0	
Thrusters	4.0	
Valving & Piping	7.6	
Jet Vane Assembly	3.0	
Structures	(47.00)	(21.34)
All Compartments	38.0	
Actuator & Deployment Mechanism	9.0	
Cabling	(10.0)	(4.54)
Contingency	(17.8)	(8.08)
Total Dry Mass, Less Experiment	273.8	124.31
Experiment Allocation	(50.00)	(22.7)
Space Craft Total (dry weight)	323.8	147.01
Expendables	(56.2)	(25.51)
Fuel & Oxidizer	45.0	
ACS N_2	10.0	
Propulsion Pressurant	1.2	
GROSS SPACE CRAFT TOTAL (LOADED)	380.00	172.52

MASS BREAKDOWN, AERONOMY/550S
TABLE 3-2

	lbm	kg
Data System	(13.75)	(6.24)
Tape Recorder	9.0	
Signal Conditioner	1.0	
Encoder	3.0	
Programmer	.75	
Communications System	(7.50)	(3.4)
Transmitter Hi Power	.75	
Transmitter Lo Power	.50	
Command Receiver	1.5	
Command Decoder	3.0	
Antennas	1.0	
Diplexer	.25	
Hybrid Ring	.5	
Power System	(54.00)	(24.52)
System Batteries	40.00	
Ordnance Batteries	2.0	
Solar cell array	5.0	
Battery Charger	.5	
Regulator	.5	
DC/DC Converter	.5	
DC/AC Converter	.5	
Power Distributor	5.0	
Propulsion System	(48.3)	(21.93)
Engine	6.3	
N_2 Tanks	16.0	
N_2	8.0	
Valves, Filters, Plumbing	18.0	

	lbm	kg
Attitude Control System	(8.0)	(3.6)
ACS Electronics	1.5	
Horizon Sensors (3)	3.0	
Sun Sensors	2.5	
Integrating Accelerometer	1.0	
Reaction Control System	(13.0)	(5.9)
Thrusters (2)	6.0	
Valving & Piping	4.0	
Spin Thrusters	3.0	
Cabling	(10.0)	(4.54)
Fixed Subsystem Total Weights	154.55	70.17
Contingency	(6.45)	(2.93)
Structures	(88)	(39.95)
Total Dry Mass, Less Experiment	249.00	113.05
Experiment Allocation	(40.00)	(18.16)
Space Craft Total (dry weight)	(289.00)	131.21
Variable Masses	(411)	(186.59)
Propellant	306	
Propellant Tank	105	
GROSS SPACE CRAFT TOTAL (LOADED)	700.00	317.8

MASS BREAKDOWN, 700 FROM TARGET

TABLE 3-3

	lbm	kg
Data System	(13.75)	(6.24)
Tape Recorder	9.0	
Signal Conditioner	1.0	
Encoder	3.0	
Programmer	.75	
Communications System	(7.50)	(3.4)
Transmitter Hi Power	.75	
Transmitter Lo Power	.50	
Command Receiver	1.5	
Command Decoder	3.0	
Antennas	1.0	
Diplexer	.25	
Hybrid Ring	.5	
Power System	(64.00)	(29.06)
System Batteries	50.0	
Ordnance Batteries	2.0	
Solar cell array	5.0	
Battery Charger	.5	
Regulator	.5	
DC/DC Converter	.5	
DC/AC Converter	.5	
Power Distributor	5.0	
Propulsion System	(76.3)	(34.64)
Engine	6.3	
N_2 Tanks	30.0	
N_2	15.0	
Valves, Filters, Plumbing	25.0	

	lbm	kg
Attitude Control System	(8.0)	(3.63)
ACS Electronics	1.5	
Horizon Sensors (3)	3.0	
Sun Sensors	2.5	
Integrating Accelerometer	1.0	
Reaction Control System	(13.0)	(5.9)
Thrusters (2)	6.0	
Valving & Piping	4.0	
Spin Thrusters	3.0	
Cabling	(10.0)	(4.54)
Fixed Subsystem Total Weights	192.6	87.41
Structures	(152)	(69.01)
Total Dry Mass, Less Experiment	344.6	156.42
Experiment Allocation	(50.0)	(22.7)
Space Craft Total (dry weight)	394.6	179.12
Variable Masses	(805)	(365.47)
Propellant	598	71.8
Propellant Tank	207	94.10
GROSS SPACE CRAFT TOTAL (LOADED)	1200	(544.59)

MASS BREAKDOWN, TARGET

TABLE 3-4

	lbm	kg
Data System	(13.75)	(6.25)
Tape Recorder	9.0	
Signal Conditioner	1.0	
Encoder	3.0	
Programmer	.75	
Communications System	(7.50)	(3.4)
Transmitter Hi Power	.75	
Transmitter Lo Power	.50	
Command Receiver	1.5	
Command Decoder	3.0	
Antennas	1.0	
Diplexer	.25	
Hybrid Ring	.5	
Power System	(53.25)	(24.18)
System Batteries	39.25	
Ordnance Batteries	2.0	
Solar cell array	5.0	
Battery Charger	.5	
Regulator	.5	
DC/DC Converter	.5	
DC/AC Converter	.5	
Power Distributor	5.0	
Propulsion System	(66.3)	(30.1)
Engine	6.3	
N_2 Tanks	25.0	
N_2	15.0	
Valves, Filters, Plumbing	20.0	

	lbm	kg
Attitude Control System	(7.0)	(3.18)
ACS Electronics	1.5	
Horizon Sensors (3)	2.5	
Sun Sensors	2.0	
Integrating Accelerometer	1.0	
Reaction Control System	(12.2)	(5.54)
Thrusters	6.0	
Valving & Piping	3.5	
Spin Thrusters	2.7	
Cabling	(9.0)	(4.09)
Fixed Subsystem Total Weights	169.0	76.73
Structures	(120)	(54.48)
Total Dry Mass, Less Experiment	289.0	131.21
Experiment Allocation	(40.0)	(18.16)
Space Craft Total (dry weight)	329.0	149.36
Variable Masses	(871)	(395.43)
Propellant	692	
Propellant Tank	179	
GROSS SPACE CRAFT TOTAL (LOADED)	1200	544.8

MASS BREAKDOWN, OPTIMISTIC

TABLE 3-5

	lbm	kg
Data System	(10.6)	(4.81)
Tape Recorder	6.6	
Signal Conditioner	1.0	
Encoder	3.0	
Communications System	(7.50)	(3.4)
Transmitter Hi Power	.75	
Transmitter Lo Power	.50	
Command Receiver	1.5	
Command Decoder	3.0	
Antennas	1.0	
Diplexer	.25	
Hybrid Ring	.5	
Power System	(82.30)	(37.36)
System Batteries	50.0	
Ordnance Batteries	2.0	
Solar Cell Array	23.3	
Battery Charger	.5	
Regulator	.5	
DC/DC Converter	.5	
DC/AC Converter	.5	
Power Distributor	5.0	
Propulsion System	(209.3)	(95.02)
Engine	6.3	
N_2 Tanks	168.0	
N_2	15.0	
Valves, Filters, Plumbing	20.0	

	lbm	kg
Attitude Control System	(20.0)	(9.08)
ACS Electronics	2.5	
Horizon Sensors (3)	10.5	
Yaw Gyro Compass	2.0	
Autopilot		
Rate Gyros	1.5	
Integrating Accelerometer	1.0	
Internal Sensor Housing	2.5	
Reaction Control System	(44.6)	(20.25)
Tanks	30.0	
Thrusters	4.0	
Valving & Piping	7.6	
Jet Vane Assembly	3.0	
Structures	(84.0)	(38.14)
All Compartments	75.0	
Actuator & Deployment Mechanism	9.0	
Cabling	(10.0)	(4.54)
Contingency	(14.2)	(6.45)
Total Dry Mass, Less Experiment	482.5	219.06
Experiment Allocation	(50.00)	(22.7)
Space Craft Total (dry weight)	532.5	241.76
Expendables	(667.5)	(303.04)
Fuel & Oxidizer	645.0	
ACS N_2	15	
Propulsion Pressurant	7.5	
GROSS SPACE CRAFT TOTAL (LOADED)	1200	544.8

MASS BREAKDOWN, AERONOMY/550T

TABLE 3-6

	POWER (Watts)	DUTY (Minutes)	ENERGY (Watt-minutes)	ORBITAL POWER AV. (Watts)	POWER PEAK (Watts)
COMMUNICATIONS SUBSYSTEM					
F. M. Transmitter	2.86	3	8.58		
P. M. Transmitter	0.29	88	25.2		
Command Receiver(interrogate)	0.68	3	2.0		
Command Receiver(standby)	0.33	85	28.0		
Command Decoder(interrogate)	2.5	3	7.5		
Command Decoder(standby)	0.002	85	0.17		
			71.5	0.82	6.33
DATA SUBSYSTEM					
PCM Encoder	2.0	88	176.0		
Signal Conditioning (housekeeping)	0.02	88	1.8		
Tape Recorder (playback)	2.0	3	6.0		
Tape Recorder (record)	1.0	85	85.0		
			268.8	3.06	4.02
ATTITUDE CONTROL SUBSYSTEM					
ACS Motors, Sensors, etc.	13.0	88	1144.0		
ACS Computer	0.2	88	17.6		
			1161.6	13.20	13.20
POWER SUBSYSTEM					
DC/DC Converter 80% eff					
Signal Conditioner 1.8 wm					
Tape Recorder 91.0 wm					
ACS Computer 17.6 wm					
110.4 wm			27.6		
DC/AC Inverter (85% eff)					
ACS Motors, Sensors 1144.0 wm			201.8		
Load Regulator					
3 Vav. drop, 20 w load, 28VDC	2.2	88	193.5		
			422.9	4.80	4.80
Total Requirements			1924.8	21.88	28.35
ADDITIONAL REQUIREMENTS (Sunlit Orbit Only)					
Battery (140% x 22 w)	30.8	35	1078.0	12.25	30.8
Total Power Requirement (Not including experiement)			3002.8	34.1	59.0

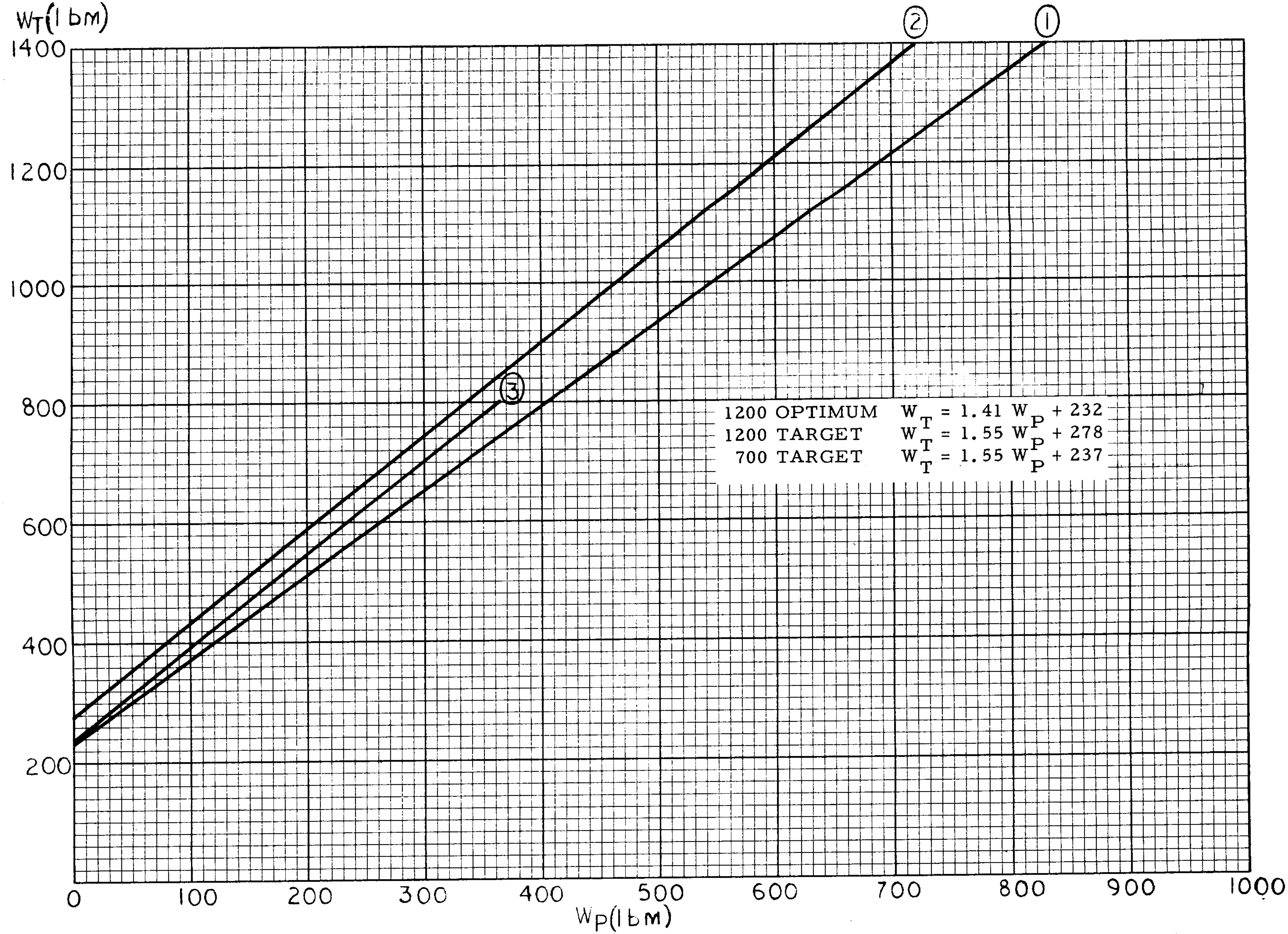
POWER REQUIREMENTS SUMMARY, AERONOMY/550

TABLE 3-7

SUBSYSTEM	POWER WATTS	DUTY (HRS)	ENERGY (WATT HRS)	S/C DAILY POWER AV. (WATTS)	TOLER. DAILY AVERAGE (WATTS)
Communications					
FM xmtr	5.75	2	11.50		
PM xmtr	0.29	24	6.95		
Command Rec. (Interr)	0.68	.5	0.34		
Command Rec. (Standby)	0.33	23.5	7.75		
Command Rec. (Interr.)	2.5	.5	1.25		
Command Rec. (Standby)	0.002	23.5	0.05		
			27.84	1.16	
Data					
PCM Encoder	4.0	2	8.0		
Signal Cond. (Housekeeping)	0.02	2	0.04		
Tape Rec. (Playback)	4.0	.2	.8		
Tape Rec. (Record)	2.0	2	4.0		
Programmer	1.0	24	24.0		
			36.84	1.54	
ACS					
ACS Motors, Sensors, etc.	5.0	2	10.0		
ACS Computer	0.2	24	4.8		
			14.8	0.62	
Power					
1. DC/DC Conv. 80% Signal Cond. .04 Tape Rec. 4.0 ACS Comp. .2 4.24	1.1	2	2.2		
2. DC/AC Inv. (85%) ACS Motors Sensors	1.8	2	3.6		
3. Load Reg.	1.0	14	14.0		
			19.8	.83	
Experiment	60 to 120	2	120 to 240	5 to 10	
TOTAL					9.15 to 14.15

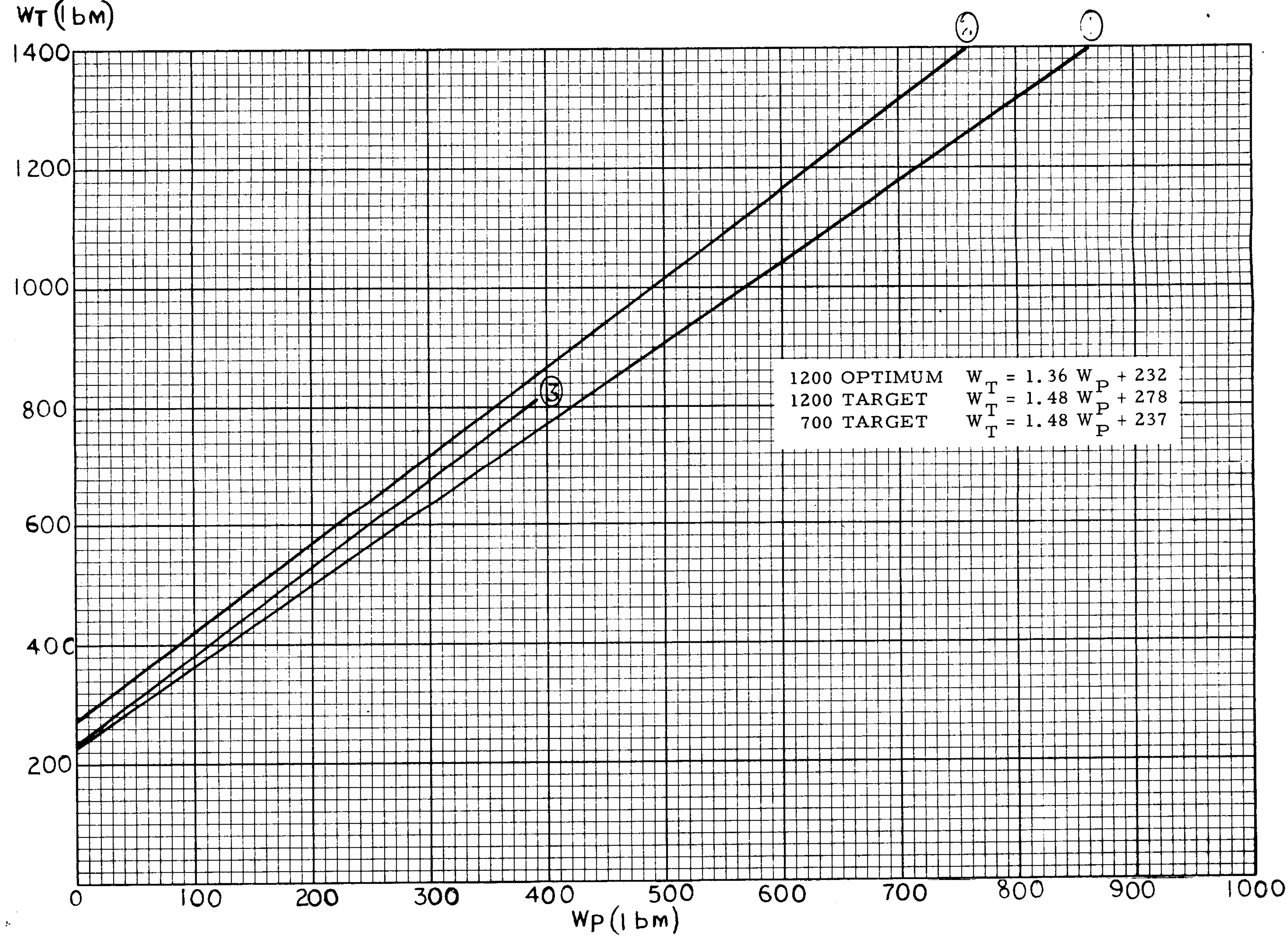
POWER REQUIREMENTS SUMMARY, AERONOMY/SS

TABLE 3-8



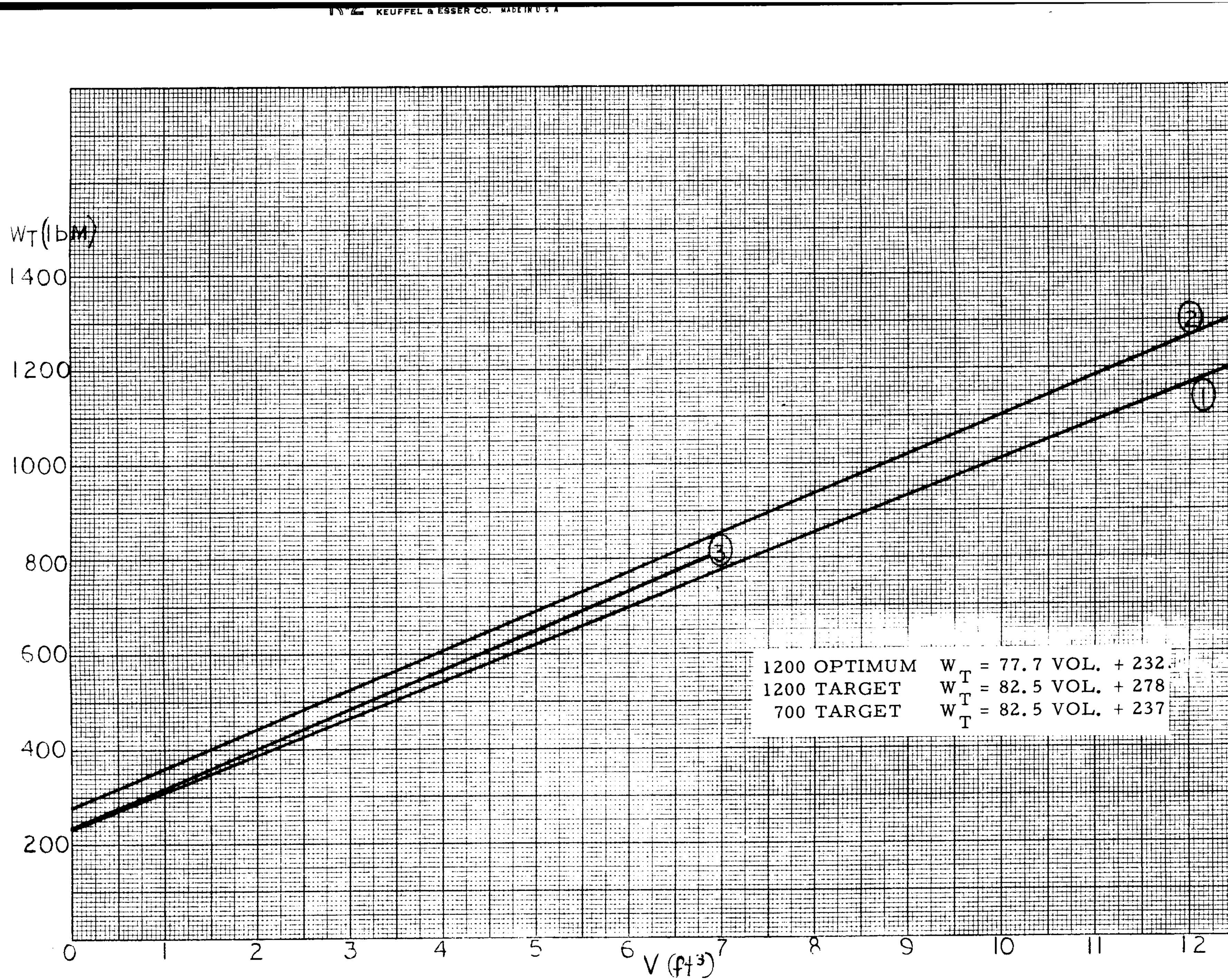
PROPELLANT MASS VS. AERONOMY/SS INITIAL MASS, HYDRAZINE SYSTEM

FIGURE 3-1



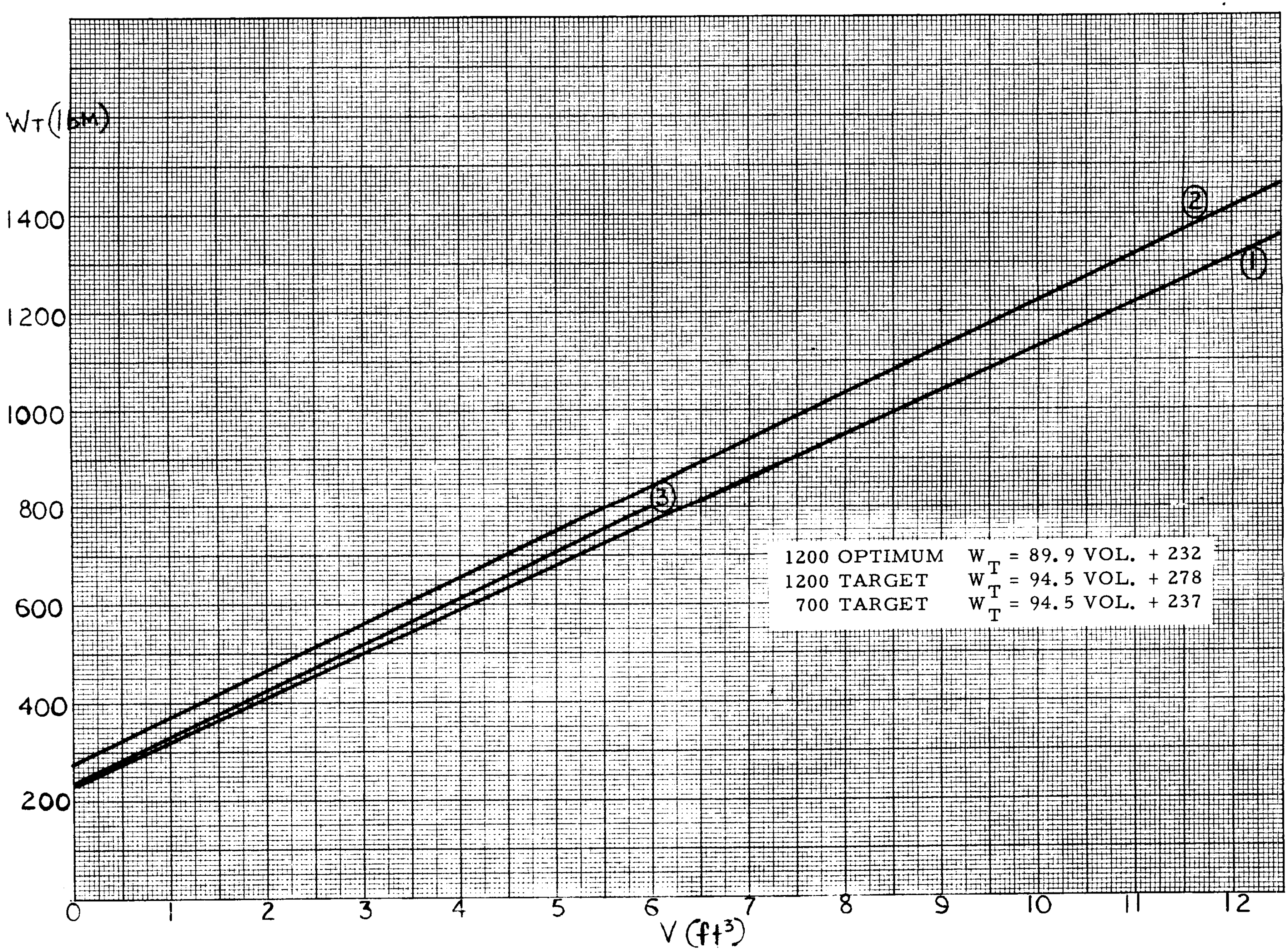
PROPELLANT MASS VS. AERONOMY/SS INITIAL MASS, BIPROPELLANT SYSTEM

FIGURE 3-2



PROPELLANT VOLUME VS. AERONOMY/SS INITIAL MASS, HYDRAZINE SYSTEM

FIGURE 3-3



PROPELLANT VOLUME VS. AERONOMY/SS INITIAL MASS, BIPROPELLANT SYSTEM

FIGURE 3-4

MISSION ANALYSIS

4.1 ORBITAL MECHANICS ANALYSIS

The aeronomy satellite experiment objectives define the bounds of the earth orbits to be considered. The desire of the experimenter to make atmospheric soundings from about 800 kilometers to 120 kilometers in altitude, within the same time frame, indicates an orbit which includes these two bounds. A one year mission time requirement can be met by two methods; (1) choosing an apogee sufficiently high to provide a one year lifetime; or (2) by putting energy back into a shorter lifetime orbit to sustain the lifetime.

The first method requires that the satellite be injected into an orbit with a 120 km perigee and about a 13,000 km apogee. This orbit would decay to a 120 km circular orbit in about a year assuming aerodynamic drag as the only perturbation. The initial period would be 4.1 hours with about 19 minutes spent below 800 km, and 8 minutes below 240 km per orbit. The initial velocity at perigee (also required injection velocity) is 31,488 ft/sec. (9,598 m/sec). In this orbit the time below 240 km is 48 minutes/day initially and about 400 minutes/day toward the end of the years mission, for a total time of about 750 hours below 240 km for the mission.

The alternative method is to use a propulsion module incorporated within the satellite to sustain the lifetime and thus permit longer mission life in the higher aerodynamic drag regions. This approach permits greater flexibility in mission design and also permits the experimenter to change the orbital parameters, at his wish, anytime during the mission.

The initial concept for the propelled mission was to inject into an 800 km to 250 km parking orbit. At intervals the perigee of the orbit would be lowered to about 120 km. With a reasonable margin of lifetime remaining the perigee would be raised and the spacecraft placed back into the parking orbit. With this approach extended lifetime is achieved by spending most of the mission time in the relatively long lifetime parking orbit and "dipping" to the lower altitudes for atmosphere soundings when desired.

4.1.1 Basic Mission Definition

In order to provide a point of departure and establish a criteria for comparing propelled satellite designs a basic mission must be defined.

Experiment requirements indicate that low altitude sweeps must be performed at least 25 times during a years mission, with a minimum of 10 orbits per low altitude sweep. On the basis of these minimum requirements a basic mission is defined as follows:

a. Inject into parking orbit (800 km apogee; 250 km perigee)
b. Remain in parking orbit for 15 days
c. Lower perigee to 120 km
d. Make 15 orbits with 120 km perigee
e. Raise perigee back to 250 km
f. Restore apogee to 800 km
g. Repeat the sequence b through f 25 times.

Steps "b" through "f" constitute a basic maneuver and 25 maneuvers make up a basic mission. A basic maneuver consist of deboosting from the parking orbit to a 120 km perigee, raising the perigee back to the parking orbit value, and restoring the parking orbit apogee. Thus, the basic maneuver requires three propulsive burns. A basic maneuver is depicted in Figure 2-1. Fifteen low sweep orbits were choosen because a lesser number would not provide a full days coverage.

The parking orbit period is about 95 minutes, and the period of the low sweep orbit is about 93.6 minutes.

4.1.2 Launch Vehicle Consideration

Launch vehicles of the Thor-Delta series are prime candidates for the Aeronomy Satellite; specifically the Thor-Delta and the Thrust Augmented Thor-Delta. These vehicles have payload capabilities of 740 and 1250 lbm into a 250 km to 800 km, 62.5° inclination orbit respectively. These payload weights delineate two possible classes of Aeronomy satellites, a 700 lbm range and a 1200 lbm range. Preliminary designs with these weights have been developed in Section 5. The performance of these designs is analyzed in Section 4.1.3.

4.1.3 Mission Analysis

With the basic mission and basic maneuver definition given in section 4.1.1 a computational procedure may be developed to determine the performance of particular satellite design. The computation procedure is best described by means of a calculation sequence and is presented in Figure 4-1. Basically, all that is entailed, is evaluating the energy loss of the satellite due to aerodynamic drag, and adding the energy required to transfer between the parking orbit and the low sweep orbit. Sequential calculations must be performed since the satellites ballistic coefficient changes after each propulsive phase (and attitude changes if a mass expulsion attitude control system is used).

A computer program was developed utilizing the computation sequence given in Figure 4-1. Lifetime curves and information on satellite decay was obtained from NASA SP33 Part I "Orbital Flight Handbook" in which the 1959 ARDC atmosphere was employed. In this publication the drag curves were obtained by numerically integrating the equations of orbital motion. Average rates of apogee and perigee decay were entered into the computer program and converted into terms of energy loss by means of the equations of orbital mechanics.

CALCULATION PROCEDURE FOR DETERMINING VEHICLE PERFORMANCE

1. Enter with initial spacecraft weight and orbit parameters.
2. Calculate the ballistic coefficient.
3. Calculate the apogee and perigee of the parking orbit after 15 days of aerodynamic decay.
4. Calculate propellant required for positioning spacecraft to proper attitude for lowering perigee.
5. Calculate new weight.
6. Calculate ΔV and propellant required to lower perigee.
7. Calculate new weight.
8. Calculate propellant to change spacecraft attitude to cruise position.
9. Calculate new weight and ballistic coefficient.
10. Calculate apogee and perigee of low sweep orbit after 15 sweeps.
11. Calculate propellant required for positioning spacecraft to proper attitude for raising perigee.
12. Calculate new weight.
13. Calculate ΔV and propellant required to restore perigee to parking orbit value.
14. Calculate new weight.
15. Calculate propellant to position spacecraft for restoration of parking orbit apogee.
16. Calculate ΔV and propellant required to restore apogee.
17. Calculate new weight.
18. Calculate propellant required to reestablish cruise attitude.
19. Calculate new weight.
20. Loop back to 2 and continue iterations until the propellant load is depleted.

CALCULATION PROCEDURE FOR DETERMINING VEHICLE PERFORMANCE

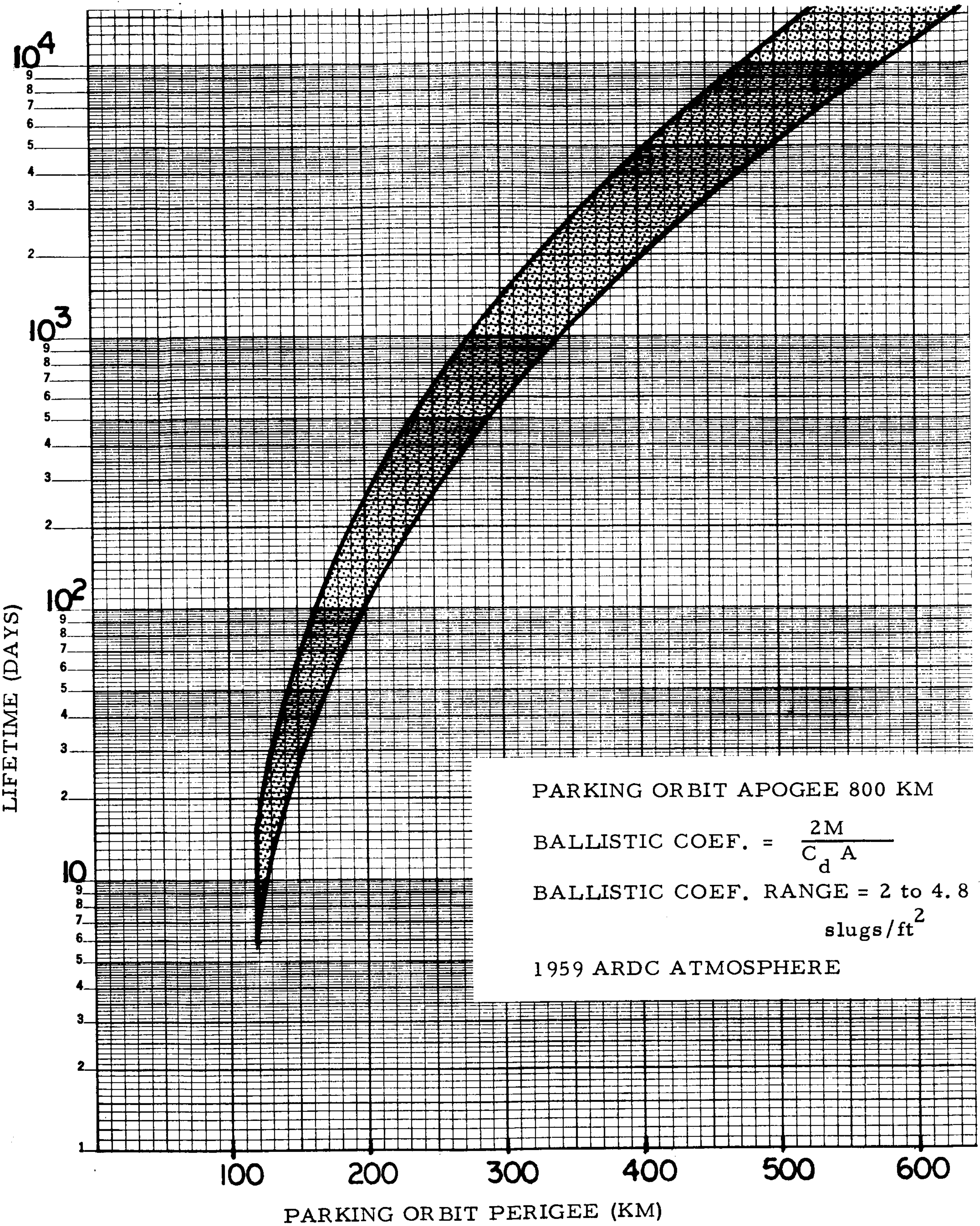
FIGURE 4-1

The five basic Aeronomy Satellites designs were evaluated by means of the computer program and the results are presented in tabular form in Tables 4-1a through 4-5c. The nomenclature employed in the tables is given in Table 4-1.

In Table 4-1a through Table 4-1f the Aeronomy/"1200" Target vehicle, as described in Table 3-4, is evaluated. Tables a through c are for a monopropellant system and d through f are for bipropellant system. A "two-pulse" mass expulsion attitude control system as described in section 5.2.7 was assumed with an effective specific impulse of 150. Mission performance was evaluated for the basic mission, and alternate missions with 230 and 200 km perigee parking orbits. Similarly Tables 4-2 evaluate the Aeronomy/"1200" Optimum, Tables 4-3 the Aeronomy/"700", Tables 4-4 the Aeronomy/"550" T, and Tables 4-5 the Aeronomy/"550"S satellites. It should be noted that the Aeronomy/"550"T and /"550"S are not spin stabilized and a gaseous N_2 limit cycle type Attitude Control System was assumed.

It was noted from the computer runs that the energy disipated due to aerodynamic drag, i.e. the energy required to restore the apogee of the parking orbit, was only about 15 percent of the total energy required for the basic mission. The major amount of energy is expended in transferring between the parking orbit and the low sweep orbit. In order to minimize this energy the distance between the parking orbit perigee and the low sweep orbit perigee can be reduced. This of course can be done only to the extent that total lifetime is not compromised.

Figure 4-2 shows that the perigee of the parking orbit can be reduced to about 200 km and a parking orbit lifetime of about 100 days is still assured.



PARKING ORBIT LIFETIME
FIGURE 4-2

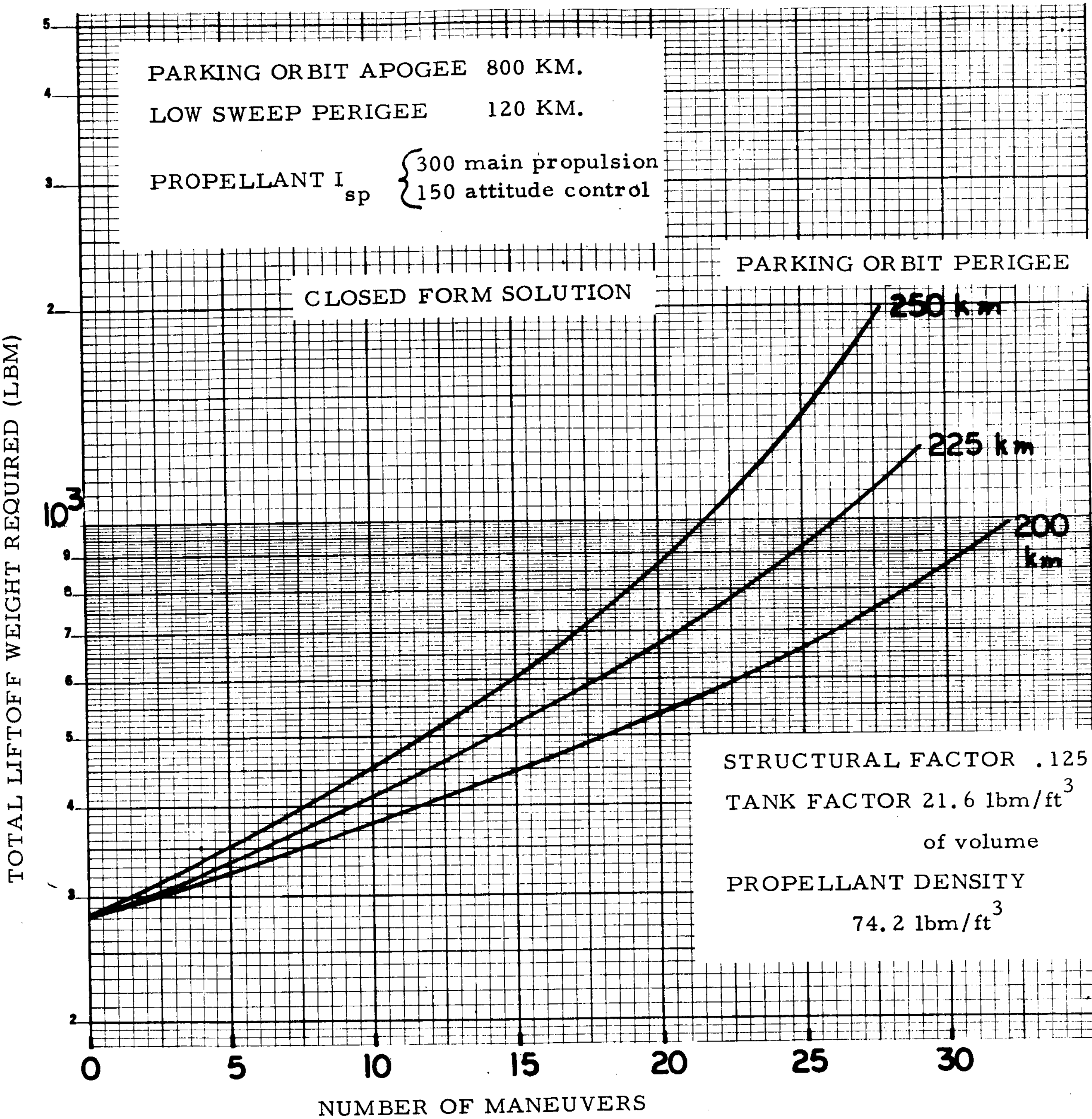
The mission tables show that reducing the parking orbit perigee results in increased total lifetime and that now approximately 30 percent of the total energy expended is used in overcoming aerodynamic drag.

These results could lead one to believe that if all of the energy available was used in overcoming the energy lost in decay, that the lifetime would be maximized. A cursory look revealed that this in general is not true. For example, if the satellite were injected into the low sweep orbit (120 km perigee, 800 km apogee) and allowed to decay to a 120 km circular orbit before restoring apogee, about 100 days of lifetime could be achieved with the "1200" Target vehicle. Whereas, Table 4-1C shows that 345 days total lifetime can be achieved with a 200 km perigee parking orbit type approach. Obviously, there is a class of orbits somewhere between these bounds where lifetime is maximized and further study is indicated.

Closed Form Solution

If the parking orbit type approach is used, it is possible to develop a closed form solution to determine the total life-off weight required to perform a given number of maneuvers. A closed form solution can be developed as a consequence of being able to parameterize satellite weights as a function of propellant weight; and, the relative independence of drag in the parking orbit type approach. The closed form solution is developed in Appendix A. Figures 4-3 and 4-4 show the effect of number of maneuvers on the total weight as a function of parking orbit perigee. The constants used in developing these figures are given in section 5.2.1 and were substituted into the equations of Appendix A.

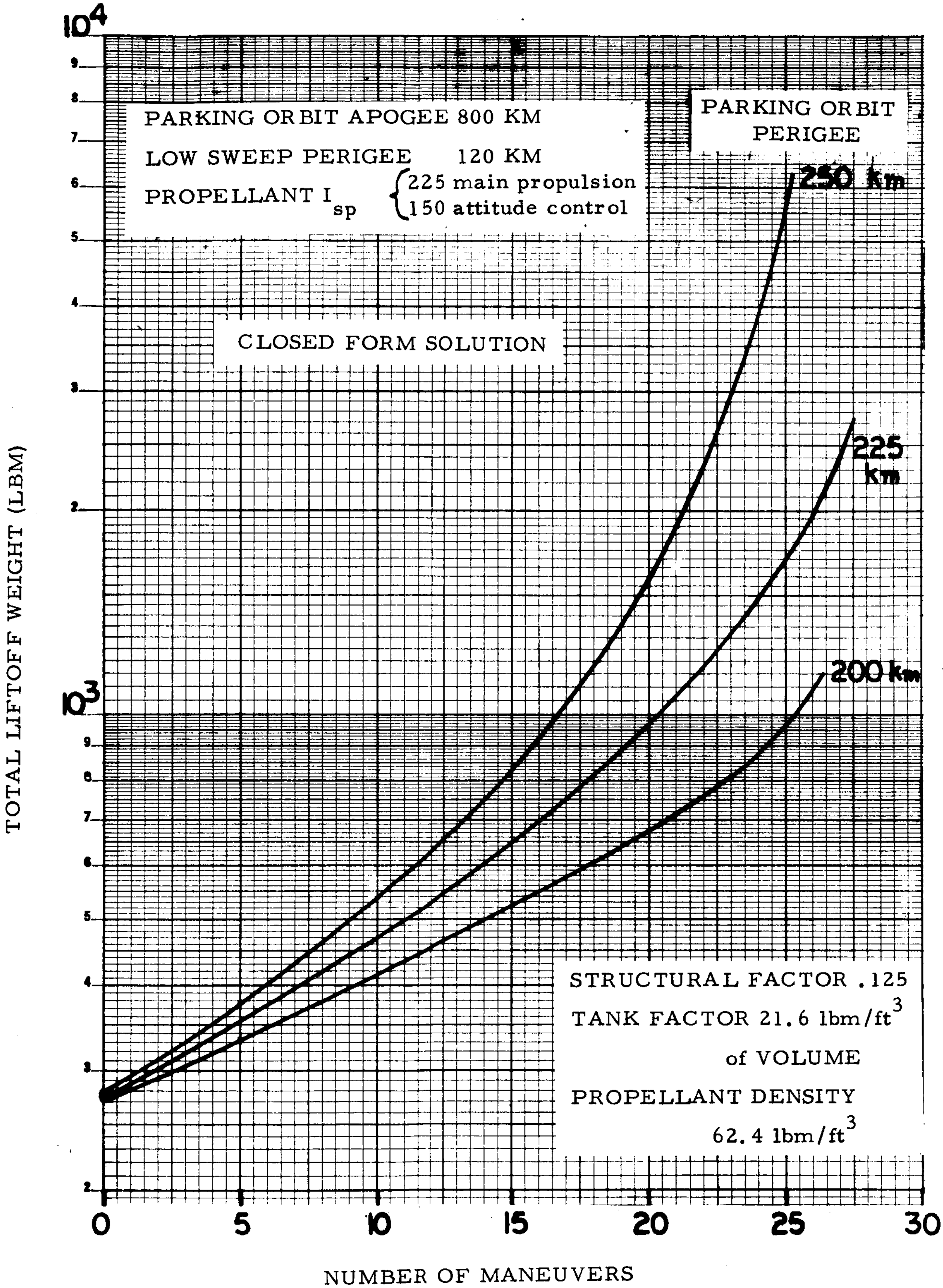
It should be noted that as the parking orbit perigee is lowered the effective error (due to neglecting the effects of drag) is increased in the closed



REQUIRED LIFT-OFF WEIGHT VS. NUMBER OF MANEUVERS

BIPROPELLANT SYSTEM

FIGURE 4-3



REQUIRED LIFT-OFF WEIGHT VS. NUMBER OF MANEUVERS

MONOPROPELLANT SYSTEM

FIGURE 4-4

form solution. The closed form solution is of value in asmuchas it provides deeper insight into the relationship between parameters.

It is interesting to note that the curves of the figures become asymptotic at some value of N. That is, an infinite total liftoff weight provides only a finite number of maneuvers. This is the familiar rocket problem wherein the mass ratio approaches a constant and no better performance can be achieved. In launch vehicles the solution to this problem is staging, and likewise, in the case of the Aeronomy Satellite if a large number of maneuvers is desired staging should be considered.

4.1.4 Principal Findings

Table 4-6 synopsizes the results of the mission tables. It is seen that any of the 1200 lbm vehicles is capable of providing a complete basic mission with the proper combination of propellant and parking orbit perigee. Only the Aeronomy/"1200" Optimum can provide a basic mission with a mono-propellant, however.

It is interesting to note that there is not really a significant difference between the 700 lbm vehicle and the 1200 lbm target vehicle. Only some 45 days to 60 days of additional lifetime at the expense of 500 lbm additional lift-off weight. Obviously a trade may be made here between additional booster cost and mission life, or extended booster capability and mission life. The reason for the seemly severe weight penalty that must be paid for the additional 45 to 60 days lifetime is the result of the exponential nature of the curves in Figures 4-3 and 4-4. The expontial form is the result of approaching a constant mass ratio as the total weight is increased.

NOMENCLATURE USED IN MISSION TABLES

MNVR	-	designates maneuver number
HA1	-	apogee of parking orbit just prior to lowering of perigee (KM)
DVA1	-	velocity increment required to lower perigee to low sweep value (ft/sec)
HA2	-	apogee of low sweep orbit after designated number of sweeps (orbits) (KM)
HP2	-	perigee of low sweep orbit after designated number of sweeps (KM)
DVA2	-	velocity increment required to raise perigee from low sweep value to parking orbit value (ft/sec)
DVP2	-	velocity increment required to restore apogee to parking orbit value (this is also a measure of the energy used to counteract drag) (ft/sec)
TDV	-	total velocity increment required for the maneuver (ft/sec)
WPACS	-	total propellant weight required for attitude changes during the maneuver (lbm)
WPT	-	total propellant used for the maneuver, including that used by the Attitude Control System and the main propulsion unit (lbm)
W	-	weight of the spacecraft at the end of the maneuver being considered (lbm)

NOMENCLATURE USED IN MISSION TABLES

TABLE 4-1

PARKING ORBIT 250.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 5.50SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 225. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	791.30	249.68	123.57	769.82	119.87	124.10	26.30	273.97	2.99	47.48	1152.52
2	790.94	249.66	123.56	768.57	119.86	124.11	27.39	275.05	2.87	93.25	1106.75
3	790.56	249.65	123.54	767.27	119.86	124.13	28.52	276.19	2.76	137.36	1062.64
4	790.17	249.63	123.53	765.91	119.85	124.14	29.71	277.38	2.65	179.89	1020.11
5	789.76	249.62	123.52	764.49	119.84	124.15	30.95	278.62	2.54	220.88	979.12
6	789.33	249.60	123.51	763.01	119.84	124.17	32.25	279.92	2.44	260.39	939.61
7	788.89	249.59	123.49	761.45	119.83	124.18	33.61	281.28	2.34	298.48	901.52
8	788.42	249.57	123.48	759.82	119.82	124.19	35.04	282.71	2.25	335.19	864.81
9	787.92	249.55	123.46	758.12	119.82	124.21	36.53	284.20	2.16	370.58	829.42
10	787.41	249.53	123.45	756.33	119.81	124.23	38.09	285.77	2.07	404.69	795.31
11	786.87	249.51	123.43	754.46	119.80	124.24	39.74	287.41	1.98	437.58	762.42
12	786.30	249.49	123.42	752.49	119.79	124.26	41.46	289.13	1.90	469.27	730.73
13	785.71	249.47	123.40	750.43	119.78	124.28	43.26	290.94	1.82	499.83	700.17
14	785.08	249.45	123.38	748.27	119.77	124.30	45.16	292.84	1.74	529.28	670.72
15	784.43	249.42	123.36	746.00	119.76	124.32	47.15	294.83	1.67	557.67	642.33
16	783.74	249.40	123.34	743.61	119.75	124.34	49.25	296.93	1.60	585.03	614.97
17	783.02	249.37	123.32	741.10	119.74	124.37	51.45	299.13	1.53	611.41	588.59
							635.86	4846.3			

Propellant Load: 598 lbm

$I_{sp} = 225$

AERONOMY/"1200" TARGET
TABLE 4-1A

PARKING ORBIT 230.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 5.50SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 225. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	789.31	229.53	104.57	767.88	119.87	105.23	28.01	237.81	3.00	41.71	1158.29
2	788.92	229.51	104.55	766.73	119.86	105.24	29.02	238.81	2.90	82.13	1117.87
3	788.52	229.49	104.54	765.52	119.86	105.25	30.07	239.86	2.80	121.29	1078.71
4	788.11	229.47	104.52	764.27	119.85	105.26	31.17	240.95	2.70	159.24	1040.76
5	787.67	229.46	104.51	762.97	119.85	105.27	32.31	242.08	2.60	196.01	1003.99
6	787.22	229.44	104.49	761.61	119.84	105.28	33.49	243.27	2.51	231.64	968.36
7	786.75	229.41	104.47	760.20	119.84	105.29	34.73	244.50	2.42	266.16	933.84
8	786.26	229.39	104.45	758.73	119.83	105.31	36.02	245.78	2.34	299.61	900.39
9	785.75	229.37	104.43	757.20	119.82	105.32	37.36	247.11	2.25	332.03	867.97
10	785.22	229.35	104.41	755.60	119.82	105.33	38.76	248.51	2.17	363.43	836.57
11	784.66	229.32	104.39	753.93	119.81	105.35	40.22	249.96	2.09	393.87	806.13
12	784.08	229.30	104.37	752.19	119.80	105.36	41.75	251.48	2.02	423.35	776.65
13	783.48	229.27	104.35	750.38	119.80	105.37	43.34	253.06	1.94	451.93	748.07
14	782.85	229.24	104.33	748.48	119.79	105.39	45.00	254.72	1.87	479.61	720.39
15	782.19	229.21	104.30	746.50	119.78	105.41	46.74	256.45	1.80	506.44	693.56
16	781.50	229.18	104.28	744.43	119.77	105.42	48.56	258.26	1.73	532.44	667.56
17	780.78	229.15	104.25	742.27	119.76	105.44	50.46	260.15	1.67	557.63	642.37
18	780.03	229.12	104.22	740.00	119.75	105.46	52.45	262.13	1.61	582.03	617.97
19	779.24	229.08	104.19	737.63	119.74	105.48	54.53	264.20	1.54	605.68	594.32

Propellant Load: 598 lbm

$I_{sp} = 225$

AERONOMY/"1200" TARGET
TABLE 4-1B

PARKING ORBIT 200.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 5.50SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 225. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	777.62	199.30	75.96	756.28	119.87	76.81	38.20	190.98	3.02	34.20	1165.80
2	776.97	199.28	75.94	755.00	119.86	76.82	39.33	192.09	2.93	67.60	1132.40
3	776.29	199.26	75.93	753.67	119.86	76.83	40.49	193.25	2.85	100.22	1099.78
4	775.58	199.24	75.91	752.30	119.86	76.84	41.70	194.44	2.76	132.08	1067.92
5	774.85	199.22	75.89	750.87	119.85	76.85	42.95	195.68	2.68	163.19	1036.81
6	774.10	199.19	75.87	749.40	119.85	76.86	44.24	196.96	2.61	193.57	1006.43
7	773.32	199.17	75.85	747.87	119.84	76.86	45.58	198.29	2.53	223.24	976.76
8	772.51	199.14	75.82	746.29	119.84	76.87	46.97	199.67	2.46	252.22	947.78
9	771.67	199.12	75.80	744.65	119.83	76.88	48.42	201.10	2.38	280.51	919.49
10	770.79	199.09	75.78	742.94	119.83	76.89	49.91	202.59	2.31	308.15	891.85
11	769.89	199.06	75.76	741.18	119.82	76.90	51.47	204.13	2.24	335.14	864.86
12	768.95	199.03	75.73	739.34	119.82	76.92	53.08	205.73	2.17	361.50	838.50
13	767.97	199.00	75.70	737.43	119.81	76.93	54.76	207.39	2.11	387.24	812.76
14	766.96	198.97	75.68	735.45	119.81	76.94	56.51	209.12	2.04	412.38	787.62
15	765.90	198.94	75.65	733.39	119.80	76.95	58.32	210.92	1.98	436.93	763.07
16	764.81	198.91	75.62	731.25	119.79	76.96	60.21	212.79	1.92	460.91	739.09
17	763.67	198.87	75.59	729.02	119.79	76.98	62.17	214.74	1.86	484.32	715.68

Propellant Load: 598 lbm

$I_{sp} = 225$

AERONOMY/"1200" TARGET

TABLE 4-1C

18	762.48	198.83	75.56	726.70	119.78	76.99	64.22	216.77	1.80	507.19	692.81
19	761.24	198.79	75.53	724.28	119.77	77.00	66.36	218.89	1.74	529.53	670.47
20	759.95	198.75	75.49	721.75	119.76	77.02	68.58	221.09	1.68	551.34	648.66
21	758.60	198.71	75.46	719.12	119.76	77.04	70.91	223.40	1.63	572.64	627.36
22	757.20	198.67	75.42	716.38	119.75	77.05	73.33	225.80	1.58	593.44	606.56
23	755.73	198.62	75.38	713.51	119.74	77.07	75.86	228.31	1.52	613.76	586.24
							1253.57	4764.13			

TABLE 4-1C (Continued)

PARKING ORBIT 250.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 5.50SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	791.30	249.68	123.57	769.91	119.87	124.10	26.22	273.89	3.01	36.52	1163.48
2	791.02	249.67	123.56	768.96	119.86	124.11	27.04	274.71	2.92	72.03	1127.97
3	790.74	249.66	123.55	767.99	119.86	124.12	27.90	275.57	2.83	106.55	1093.45
4	790.45	249.65	123.54	766.97	119.86	124.13	28.78	276.45	2.74	140.11	1059.89
5	790.15	249.63	123.53	765.93	119.85	124.14	29.70	277.37	2.66	172.74	1027.26
6	789.83	249.62	123.52	764.85	119.85	124.15	30.64	278.31	2.58	204.46	995.54
7	789.51	249.61	123.51	763.73	119.84	124.16	31.62	279.29	2.50	235.30	964.70
8	789.17	249.60	123.50	762.57	119.84	124.17	32.64	280.31	2.42	265.28	934.72
9	788.83	249.59	123.49	761.37	119.83	124.18	33.69	281.36	2.34	294.43	905.57
10	788.47	249.57	123.48	760.12	119.83	124.19	34.77	282.45	2.27	322.77	877.23
11	788.10	249.56	123.47	758.83	119.82	124.20	35.90	283.57	2.20	350.32	849.68
12	787.71	249.54	123.46	757.50	119.81	124.22	37.07	284.74	2.13	377.10	822.90
13	787.31	249.53	123.45	756.12	119.81	124.23	38.28	285.95	2.06	403.14	796.86
14	786.89	249.51	123.43	754.68	119.80	124.24	39.54	287.21	2.00	428.46	771.54
15	786.46	249.50	123.42	753.20	119.80	124.25	40.84	288.52	1.93	453.07	746.93
16	786.02	249.48	123.41	751.65	119.79	124.27	42.19	289.87	1.87	477.00	723.00
17	785.56	249.46	123.39	750.05	119.78	124.28	43.59	291.27	1.81	500.27	699.73

Propellant Load: 598 lbm

$I_{sp} = 300$

AERONOMY/"1200" TARGET

TABLE 4-1D

18	785.08	249.45	123.38	748.39	119.77	124.30	45.05	292.73	1.75	522.89	677.11
19	784.58	249.43	123.36	746.67	119.77	124.31	46.56	294.24	1.70	544.88	655.12
20	784.06	249.41	123.35	744.88	119.76	124.33	48.13	295.81	1.64	566.25	633.75
21	783.52	249.39	123.33	743.02	119.75	124.35	49.76	297.44	1.59	587.04	612.96
22	782.96	249.37	123.32	741.09	119.74	124.37	51.46	299.14	1.54	607.25	592.75

TABLE 4-1D (Continued)

PARKING ORBIT 230.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 5.50SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	789.31	229.53	104.57	767.96	119.87	105.23	27.94	237.74	3.02	32.16	1167.84
2	789.01	229.51	104.56	767.08	119.87	105.24	28.71	238.51	2.94	63.55	1136.45
3	788.71	229.50	104.55	766.17	119.86	105.25	29.51	239.30	2.86	94.18	1105.82
4	788.40	229.49	104.53	765.23	119.86	105.25	30.33	240.11	2.78	124.08	1075.92
5	788.08	229.47	104.52	764.26	119.85	105.26	31.17	240.95	2.70	153.27	1046.73
6	787.74	229.46	104.51	763.27	119.85	105.27	32.04	241.82	2.63	181.75	1018.25
7	787.40	229.44	104.50	762.24	119.85	105.28	32.94	242.72	2.56	209.55	990.45
8	787.05	229.43	104.48	761.18	119.84	105.29	33.87	243.64	2.49	236.68	963.32
9	786.68	229.41	104.47	760.09	119.84	105.29	34.83	244.59	2.42	263.16	936.84
10	786.30	229.39	104.46	758.96	119.83	105.30	35.82	245.58	2.35	289.01	910.99
11	785.92	229.38	104.44	757.80	119.83	105.31	36.84	246.59	2.29	314.24	885.76
12	785.52	229.36	104.43	756.59	119.82	105.32	37.89	247.64	2.23	338.86	861.14
13	785.10	229.34	104.41	755.35	119.82	105.33	38.98	248.72	2.16	362.89	837.11
14	784.67	229.32	104.39	754.07	119.81	105.34	40.10	249.84	2.10	386.34	813.66
15	784.23	229.30	104.38	752.75	119.81	105.35	41.26	250.99	2.05	409.24	790.76
16	783.77	229.28	104.36	751.38	119.80	105.37	42.46	252.19	1.99	431.58	768.42
17	783.30	229.26	104.34	749.97	119.79	105.38	43.70	253.42	1.93	453.38	746.62

Propellant Load: 598 lbm

$I_{sp} = 300$

AERONOMY/"1200" TARGET

TABLE 4-1E

18	782.82	229.24	104.32	748.50	119.79	105.39	44.98	254.70	1.88	474.67	725.33
19	782.31	229.22	104.31	746.99	119.78	105.40	46.31	256.02	1.82	495.44	704.56
20	781.79	229.20	104.29	745.43	119.78	105.41	47.68	257.38	1.77	515.72	684.28
21	781.25	229.17	104.27	743.81	119.77	105.43	49.10	258.79	1.72	535.50	664.50
22	780.69	229.15	104.24	742.14	119.76	105.44	50.57	260.26	1.67	554.82	645.18
23	780.11	229.12	104.22	740.41	119.76	105.45	52.09	261.77	1.62	573.67	626.33
24	779.52	229.09	104.20	738.61	119.75	105.47	53.67	263.34	1.57	592.07	607.93
25	778.90	229.07	104.18	736.76	119.74	105.48	55.30	264.96	1.53	610.03	589.97

TABLE 4-1E (Continued)

PARKING ORBIT 200.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 5.50SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	777.62	199.30	75.96	756.34	119.87	76.81	38.15	190.93	3.03	26.49	1173.51
2	777.12	199.29	75.95	755.35	119.87	76.82	39.02	191.78	2.96	52.49	1147.51
3	776.60	199.27	75.93	754.34	119.86	76.83	39.90	192.66	2.90	78.02	1121.98
4	776.07	199.26	75.92	753.30	119.86	76.83	40.82	193.57	2.83	103.09	1096.91
5	775.52	199.24	75.90	752.23	119.86	76.84	41.75	194.50	2.77	127.69	1072.31
6	774.96	199.22	75.89	751.14	119.85	76.84	42.72	195.45	2.71	151.85	1048.15
7	774.38	199.20	75.87	750.01	119.85	76.85	43.70	196.43	2.64	175.57	1024.43
8	773.79	199.18	75.86	748.85	119.85	76.86	44.72	197.44	2.58	198.86	1001.14
9	773.18	199.17	75.84	747.66	119.84	76.87	45.77	198.47	2.53	221.72	978.28
10	772.55	199.15	75.83	746.44	119.84	76.87	46.84	199.54	2.47	244.17	955.83
11	771.91	199.13	75.81	745.18	119.84	76.88	47.95	200.64	2.41	266.20	933.80
12	771.24	199.11	75.79	743.89	119.83	76.89	49.08	201.76	2.36	287.84	912.16
13	770.56	199.08	75.77	742.56	119.83	76.90	50.25	202.92	2.30	309.08	890.92
14	769.86	199.06	75.75	741.19	119.82	76.90	51.46	204.12	2.25	329.93	870.07
15	769.14	199.04	75.74	739.78	119.82	76.91	52.70	205.35	2.20	350.41	849.59
16	768.39	199.02	75.72	738.33	119.81	76.92	53.97	206.61	2.14	370.51	829.49
17	767.63	198.99	75.70	736.83	119.81	76.93	55.29	207.91	2.09	390.24	809.76

Propellant Load: 598 lbm

$I_{sp} = 300$

AERONOMY/"1200" TARGET

TABLE 4-1F

18	766.84	198.97	75.67	735.29	119.81	76.94	56.64	209.26	2.04	409.61	790.39
19	766.02	198.94	75.65	733.71	119.80	76.95	58.04	210.64	1.99	428.64	771.36
20	765.19	198.92	75.63	732.07	119.80	76.96	59.48	212.07	1.95	447.31	752.69
21	764.32	198.89	75.61	730.39	119.79	76.97	60.97	213.54	1.90	465.65	734.35
22	763.43	198.86	75.58	728.65	119.79	76.98	62.50	215.06	1.85	483.65	716.35
23	762.51	198.83	75.56	726.86	119.78	76.99	64.08	216.63	1.81	501.32	698.68
24	761.56	198.80	75.53	725.01	119.78	77.00	65.71	218.25	1.76	518.67	681.33
25	760.59	198.77	75.51	723.10	119.77	77.01	67.40	219.92	1.72	535.71	654.29
26	759.58	198.74	75.48	721.13	119.76	77.02	69.14	221.64	1.68	552.43	647.57
27	758.53	198.71	75.45	719.09	119.76	77.04	70.93	223.42	1.63	568.85	631.15
28	757.45	198.68	75.42	716.98	119.75	77.05	72.79	225.27	1.59	584.97	615.03
29	756.34	198.64	75.39	714.81	119.74	77.06	74.72	227.17	1.55	600.80	599.20

TABLE 4-1F (Continued)

PARKING ORBIT 250.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 6.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 225. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	790.51	249.65	123.54	767.07	119.86	124.13	28.70	276.37	2.99	47.86	1152.14
2	790.11	249.63	123.53	765.70	119.85	124.14	29.89	277.56	2.87	93.99	1106.01
3	789.70	249.62	123.52	764.27	119.84	124.15	31.14	278.81	2.76	138.46	1061.54
4	789.27	249.60	123.50	762.78	119.84	124.17	32.45	280.12	2.65	181.33	1018.67
5	788.82	249.58	123.49	761.21	119.83	124.18	33.82	281.50	2.54	222.65	977.35
6	788.34	249.57	123.48	759.57	119.82	124.20	35.26	282.93	2.44	262.48	937.52
7	787.85	249.55	123.46	757.85	119.82	124.21	36.76	284.44	2.34	300.88	899.12
8	787.33	249.53	123.45	756.05	119.81	124.23	38.34	286.01	2.24	337.88	862.12
9	786.79	249.51	123.43	754.17	119.80	124.25	39.99	287.67	2.15	373.56	826.44
10	786.21	249.49	123.41	752.19	119.79	124.26	41.72	289.40	2.06	407.95	792.05
11	785.62	249.47	123.39	750.11	119.78	124.28	43.54	291.22	1.97	441.09	758.91
12	784.99	249.44	123.38	747.93	119.77	124.30	45.45	293.13	1.89	473.05	726.95
13	784.33	249.42	123.36	745.65	119.76	124.32	47.46	295.14	1.81	503.85	696.15
14	783.63	249.39	123.34	743.24	119.75	124.35	49.57	297.25	1.73	533.53	666.47
15	782.91	249.37	123.31	740.71	119.74	124.37	51.79	299.47	1.66	562.15	637.85
16	782.14	249.34	123.29	738.05	119.73	124.39	54.13	301.81	1.59	589.74	610.26
17	781.33	249.31	123.27	735.25	119.72	124.42	56.59	304.27	1.52	616.33	583.67
18	780.48	249.28	123.24	732.30	119.70	124.45	59.18	306.87	1.45	641.97	558.03
19	779.58	249.24	123.21	729.19	119.69	124.47	61.92	309.60	1.39	666.67	533.33
20	778.64	249.21	123.18	725.91	119.68	124.50	64.80	312.49	1.33	690.49	509.51

Propellant Load: 695 lbm

$I_{sp} = 225$

AERONOMY/"1200" OPTIMUM

TABLE 4-2A

PARKING ORBIT 230.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 6.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 225. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	788.34	229.48	104.53	764.96	119.86	105.26	30.56	240.35	3.00	42.12	1157.88
2	787.91	229.47	104.52	763.69	119.85	105.27	31.68	241.46	2.90	82.93	1117.07
3	787.47	229.45	104.50	762.36	119.85	105.28	32.84	242.61	2.79	122.47	1077.53
4	787.01	229.43	104.48	760.98	119.84	105.29	34.05	243.82	2.70	160.79	1039.21
5	786.53	229.40	104.46	759.54	119.83	105.30	35.31	245.07	2.60	197.92	1002.08
6	786.03	229.38	104.45	758.04	119.83	105.31	36.62	246.38	2.51	233.89	966.11
7	785.51	229.36	104.43	756.48	119.82	105.32	37.99	247.74	2.42	268.75	931.25
8	784.97	229.34	104.41	754.85	119.81	105.34	39.42	249.16	2.33	302.53	897.47
9	784.40	229.31	104.38	753.15	119.81	105.35	40.90	250.64	2.24	335.26	864.74
10	783.81	229.28	104.36	751.38	119.80	105.37	42.46	252.19	2.16	366.97	833.03
11	783.20	229.26	104.34	749.53	119.79	105.38	44.08	253.80	2.08	397.70	802.30
12	782.55	229.23	104.31	747.60	119.78	105.40	45.78	255.49	2.01	427.48	772.52
13	781.88	229.20	104.29	745.58	119.78	105.41	47.55	257.26	1.93	456.33	743.67
14	781.18	229.17	104.26	743.46	119.77	105.43	49.41	259.10	1.86	484.29	715.71
15	780.44	229.14	104.24	741.26	119.76	105.45	51.35	261.03	1.79	511.38	688.62
16	779.67	229.10	104.21	738.95	119.75	105.47	53.38	263.05	1.72	537.63	662.37
17	778.87	229.07	104.18	736.53	119.74	105.49	55.50	265.17	1.66	563.06	636.94
18	778.03	229.03	104.14	733.99	119.73	105.51	57.73	267.39	1.59	587.70	612.30
19	777.14	228.99	104.11	731.34	119.72	105.53	60.07	269.71	1.53	611.58	588.42
20	776.21	228.95	104.08	728.55	119.71	105.55	62.53	272.15	1.47	634.72	565.28
21	775.24	228.91	104.04	725.62	119.69	105.58	65.10	274.72	1.41	657.15	542.85
22	774.22	228.86	104.00	722.55	119.68	105.60	67.81	277.41	1.36	678.87	521.13
23	773.14	228.81	103.96	719.32	119.67	105.63	70.66	280.24	1.30	699.92	500.08

Propellant Load: 695 lbm
$I_{sp} = 225$

AERONOMY/"1200" OPTIMUM
TABLE 4-2B

PARKING ORBIT 200.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 6.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 225. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	775.59	199.24	75.91	752.31	119.86	76.84	41.69	194.43	3.02	34.76	1165.24
2	774.86	199.22	75.89	750.88	119.85	76.85	42.94	195.67	2.93	68.70	1131.30
3	774.10	199.19	75.87	749.41	119.85	76.86	44.23	196.95	2.84	101.85	1098.15
4	773.32	199.17	75.85	747.88	119.84	76.86	45.57	198.28	2.76	134.22	1065.78
5	772.51	199.14	75.83	746.30	119.84	76.87	46.96	199.66	2.68	165.84	1034.16
6	771.67	199.12	75.80	744.66	119.83	76.88	48.41	201.09	2.60	196.72	1003.28
7	770.80	199.09	75.78	742.96	119.83	76.89	49.90	202.58	2.52	226.87	973.13
8	769.90	199.06	75.76	741.19	119.82	76.90	51.46	204.12	2.45	256.32	943.68
9	768.96	199.03	75.73	739.35	119.82	76.92	53.07	205.72	2.37	285.08	914.92
10	767.98	199.00	75.70	737.45	119.81	76.93	54.75	207.38	2.30	313.16	886.84
11	766.97	198.97	75.68	735.47	119.81	76.94	56.49	209.11	2.23	340.59	859.41
12	765.91	198.94	75.65	733.41	119.80	76.95	58.31	210.91	2.16	367.38	832.62
13	764.82	198.91	75.62	731.26	119.79	76.96	60.20	212.78	2.09	393.54	806.46
14	763.67	198.87	75.59	729.03	119.79	76.98	62.16	214.73	2.03	419.09	780.91
15	762.49	198.83	75.56	726.71	119.78	76.99	64.21	216.76	1.96	444.04	755.96
16	761.25	198.79	75.53	724.29	119.77	77.00	66.34	218.87	1.90	468.41	731.59
17	759.96	198.75	75.49	721.77	119.77	77.02	68.57	221.08	1.84	492.21	707.79

Propellant Load: 695 lbm

$I_{sp} = 225$

AERONOMY/"1200" OPTIMUM

TABLE 4-2C

18	758.61	198.71	75.46	719.14	119.76	77.04	70.89	223.38	1.78	515.45	684.55
19	757.21	198.67	75.42	716.40	119.75	77.05	73.31	225.78	1.72	538.15	661.85
20	755.74	198.62	75.38	713.53	119.74	77.07	75.85	228.29	1.66	560.31	639.69
21	754.20	198.58	75.34	710.53	119.73	77.09	78.50	230.92	1.61	581.96	618.04
22	752.60	198.53	75.30	707.40	119.72	77.11	81.27	233.67	1.55	603.10	596.90
23	750.92	198.47	75.25	704.12	119.71	77.12	84.17	236.55	1.50	623.75	576.25
24	749.16	198.42	75.20	700.69	119.70	77.15	87.21	239.56	1.45	643.92	556.08
25	747.32	198.36	75.15	697.08	119.69	77.17	90.41	242.73	1.40	663.61	536.39
26	745.38	198.30	[illegible]	693.31	119.68	77.19	93.76	246.05	1.35	682.84	517.16
27	743.35	198.24	75.05	689.34	119.67	77.21	97.28	249.54	1.30	721.63	498.37

TABLE 4-2C (Continued)

PARKING ORBIT 250.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 6.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	790.51	249.65	123.54	767.17	119.86	124.13	28.61	276.28	3.01	36.81	1163.19
2	790.21	249.64	123.53	766.13	119.85	124.14	29.52	277.19	2.92	72.60	1127.40
3	789.89	249.62	123.52	765.06	119.85	124.15	30.46	278.13	2.83	107.39	1092.61
4	789.57	249.61	123.51	763.94	119.84	124.16	31.43	279.10	2.74	141.21	1058.79
5	789.24	249.60	123.50	762.79	119.84	124.17	32.44	280.11	2.65	174.10	1025.90
6	788.90	249.59	123.49	761.60	119.83	124.18	33.48	281.15	2.57	206.07	993.93
7	788.54	249.57	123.48	760.37	119.83	124.19	34.56	282.23	2.49	237.15	962.85
8	788.17	249.56	123.47	759.09	119.82	124.20	35.68	283.35	2.41	267.37	932.63
9	787.78	249.55	123.46	757.76	119.82	124.21	36.84	284.52	2.34	296.75	903.25
10	787.39	249.53	123.45	756.39	119.81	124.23	38.05	285.72	2.26	325.31	874.69
11	786.98	249.52	123.44	754.96	119.80	124.24	39.29	286.97	2.19	353.08	846.92
12	786.55	249.50	123.42	753.49	119.80	124.25	40.59	288.26	2.12	380.08	819.92
13	786.11	249.48	123.41	751.95	119.79	124.27	41.93	289.60	2.06	406.32	793.68
14	785.65	249.47	123.40	750.36	119.78	124.28	43.32	291.00	1.99	431.84	768.16
15	785.17	249.45	123.38	748.72	119.78	124.30	44.77	292.44	1.93	456.65	743.35
16	784.67	249.43	123.37	747.00	119.77	124.31	46.27	293.95	1.86	480.77	719.23
17	784.16	249.41	123.35	745.23	119.76	124.33	47.83	295.51	1.80	504.22	695.78

Propellant Load: 695 lbm

$I_{sp} = 300$

AERONOMY/"1200" OPTIMUM

TABLE 4-2D

18	783.63	249.39	123.34	743.38	119.75	124.34	49.45	297.13	1.74	527.01	672.99
19	783.07	249.37	123.32	741.46	119.74	124.36	51.13	298.81	1.69	549.18	650.82
20	782.50	249.35	123.30	739.47	119.74	124.38	52.88	300.56	1.63	570.73	629.27
21	781.90	249.33	123.28	737.40	119.73	124.40	54.70	302.38	1.58	591.67	608.33
22	781.27	249.30	123.26	735.24	119.72	124.42	56.60	304.28	1.52	612.04	587.96
23	780.62	249.28	123.24	733.00	119.71	124.44	58.57	306.25	1.47	631.84	568.16
24	779.95	249.26	123.22	730.66	119.70	124.46	60.62	308.31	1.42	651.09	548.91
25	779.24	249.23	123.20	728.23	119.69	124.48	62.76	310.45	1.38	669.81	530.19
26	778.51	249.20	123.18	725.70	119.67	124.51	64.99	312.68	1.33	688.01	511.99
27	777.75	249.17	123.16	723.06	119.66	124.53	67.32	315.01	1.28	705.70	494.30

TABLE 4-2D (Continued)

PARKING ORBIT 230.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 6.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	788.34	229.48	104.53	765.05	119.86	105.25	30.49	240.27	3.02	32.47	1167.53
2	788.01	229.47	104.52	764.08	119.85	105.26	31.34	241.12	2.93	64.15	1135.85
3	787.68	229.46	104.51	763.07	119.85	105.27	32.21	241.99	2.86	95.08	1104.92
4	787.33	229.44	104.49	762.04	119.84	105.28	33.12	242.89	2.78	125.27	1074.73
5	786.98	229.42	104.48	760.97	119.84	105.29	34.05	243.82	2.70	154.73	1045.27
6	786.61	229.41	104.47	759.87	119.84	105.30	35.02	244.78	2.63	183.48	1016.52
7	786.23	229.39	104.45	758.74	119.83	105.31	36.01	245.77	2.56	211.55	988.45
8	785.84	229.37	104.44	757.57	119.83	105.32	37.04	246.79	2.48	238.94	961.86
9	785.44	229.36	104.42	756.36	119.82	105.33	38.10	247.84	2.42	265.67	934.33
10	785.02	229.34	104.41	755.11	119.82	105.34	39.19	248.93	2.35	291.77	908.23
11	784.59	229.32	104.39	753.82	119.81	105.35	40.32	250.06	2.28	317.23	882.77
12	784.14	229.30	104.37	752.49	119.81	105.36	41.49	251.22	2.22	342.09	857.91
13	783.69	229.28	104.36	751.11	119.80	105.37	42.70	252.42	2.16	366.35	833.65
14	783.21	229.26	104.34	749.69	119.79	105.38	43.94	253.66	2.10	390.03	809.97
15	782.72	229.24	104.32	748.22	119.79	105.39	45.23	254.95	2.04	413.14	786.86
16	782.21	229.21	104.30	746.70	119.78	105.40	46.57	256.28	1.98	435.69	764.31
17	781.69	229.19	104.28	745.12	119.77	105.42	47.95	257.65	1.92	457.71	742.29

Propellant Load: 695 lbm

$I_{sp} = 300$

AERONOMY/"1200" OPTIMUM

TABLE 4-2E

18	781.14	229.17	104.26	743.50	119.77	105.43	49.38	259.07	1.87	479.19	720.81
19	780.58	229.14	104.24	741.81	119.76	105.44	50.86	260.54	1.81	500.17	699.83
20	780.00	229.12	104.22	740.07	119.75	105.46	52.39	262.07	1.76	520.63	679.37
21	779.40	229.09	104.20	738.26	119.75	105.47	53.98	263.65	1.71	540.61	659.39
22	778.77	229.06	104.17	736.39	119.74	105.49	55.62	265.28	1.66	560.11	639.89
23	778.13	229.03	104.15	734.45	119.73	105.50	57.33	266.98	1.61	579.14	620.86
24	777.46	229.00	104.12	732.44	119.72	105.52	59.10	268.74	1.56	597.72	602.28
25	776.76	228.97	104.10	730.36	119.71	105.54	60.93	270.56	1.51	615.85	584.15
26	776.04	228.94	104.07	728.20	119.71	105.55	62.83	272.44	1.47	633.54	566.46
27	775.29	228.91	104.04	725.96	119.70	105.57	64.81	274.42	1.42	650.81	549.19
28	774.51	228.87	104.01	723.63	119.69	105.59	66.86	276.46	1.38	667.67	532.33
29	773.71	228.84	103.98	721.21	119.68	105.61	68.99	278.59	1.34	684.12	515.88
30	772.87	228.80	103.95	718.70	119.67	105.63	71.21	280.79	1.30	700.18	499.82

TABLE 4-2E (Continued)

PARKING ORBIT 200.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 6.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300. PULSING ISP 150. SPACECRAFT RADIUS 2.50FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	775.59	199.24	75.91	752.37	119.86	76.84	41.64	194.38	3.03	26.91	1173.09
2	775.03	199.22	75.89	751.27	119.85	76.84	42.59	195.33	2.96	53.32	1146.68
3	774.45	199.21	75.88	750.15	119.85	76.85	43.58	196.31	2.89	79.26	1120.74
4	773.86	199.19	75.86	749.00	119.85	76.86	44.59	197.31	2.83	104.72	1095.28
5	773.25	199.17	75.84	747.81	119.84	76.86	45.63	198.34	2.76	129.72	1070.28
6	772.63	199.15	75.83	746.59	119.84	76.87	46.71	199.41	2.70	154.26	1045.74
7	771.99	199.13	75.81	745.34	119.84	76.88	47.81	200.50	2.64	178.35	1021.65
8	771.33	199.11	75.79	744.05	119.83	76.89	48.94	201.62	2.58	202.01	997.99
9	770.65	199.09	75.78	742.73	119.83	76.89	50.11	202.78	2.52	225.23	974.77
10	769.95	199.06	75.76	741.36	119.82	76.90	51.31	203.97	2.46	248.03	951.97
11	769.23	199.04	75.74	739.96	119.82	76.91	52.54	205.19	2.40	270.42	929.58
12	768.49	199.02	75.72	738.51	119.82	76.92	53.81	206.45	2.35	292.40	907.60
13	767.72	199.00	75.70	737.02	119.81	76.93	55.12	207.75	2.29	313.98	886.02
14	766.94	198.97	75.68	735.49	119.81	76.94	56.47	209.09	2.24	335.16	864.84
15	766.13	198.95	75.66	733.91	119.80	76.95	57.87	210.47	2.18	355.96	844.04
16	765.29	198.92	75.63	732.28	119.80	76.96	59.30	211.89	2.13	376.38	823.62
17	764.43	198.89	75.61	730.60	119.79	76.97	60.78	213.36	2.08	396.42	803.58

Propellant Load: 695 lbm

$I_{sp} = 300$

AERONOMY/"1200" OPTIMUM
TABLE 4-2F

18	763.54	198.87	75.59	728.87	119.79	76.98	62.31	214.87	2.03	416.11	783.89
19	762.63	198.84	75.56	727.08	119.78	76.99	63.88	216.43	1.98	435.43	764.57
20	761.68	198.81	75.54	725.24	119.78	77.00	65.51	218.04	1.93	454.40	745.60
21	760.71	198.78	75.51	723.34	119.77	77.01	67.18	219.71	1.88	473.03	726.97
22	759.70	198.75	75.48	721.37	119.76	77.02	68.92	221.42	1.83	491.31	708.69
23	758.66	198.71	75.46	719.35	119.76	77.03	70.71	223.20	1.79	509.27	690.73
24	757.59	198.68	75.43	717.25	119.75	77.05	72.56	225.03	1.74	526.90	673.10
25	756.48	198.65	75.40	715.08	119.75	77.06	74.47	226.93	1.70	544.20	655.80
26	755.37	[illegible]	75.37	712.84	119.74	77.07	76.46	[illegible]	1.65	561.19	638.81
27	754.14	198.57	75.34	710.52	119.73	77.09	78.51	230.93	1.61	577.87	622.13
28	752.91	198.54	75.30	708.13	119.72	77.10	80.63	233.03	1.57	594.25	605.75
29	751.64	198.50	75.27	705.64	119.72	77.12	82.83	235.21	1.53	610.33	589.67
30	750.32	198.45	75.23	703.07	119.71	77.13	85.10	237.47	1.49	626.12	573.88
31	748.95	198.41	75.20	700.40	119.70	77.15	87.47	239.81	1.45	641.62	558.38
32	747.54	198.37	75.16	697.64	119.69	77.16	89.92	242.24	1.41	656.84	543.16
33	746.07	198.32	75.12	694.77	119.68	77.18	92.46	244.76	1.37	671.78	528.22
34	744.54	198.27	75.08	691.79	119.68	77.20	95.10	247.38	1.33	686.45	513.55
35	742.96	198.23	75.04	688.70	119.67	77.22	97.84	250.10	1.29	700.85	499.15

TABLE 4-2F (Continued)

PARKING ORBIT 250.KM PERIGEE 800.KM APOGEE   LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15   DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 700.LBM   DRAG AREA 4.00SQ.FT.   DRAG COEF. 2.6

PROPELLANT ISP 225.   PULSING ISP 150.   SPACECRAFT RADIUS 2.00FT

1959 ARDC ATMOSPHERE   SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	789.15	249.60	123.50	762.37	119.84	124.17	32.81	280.48	1.40	27.96	672.04
2	788.70	249.58	123.49	760.81	119.83	124.19	34.18	281.85	1.34	54.92	645.08
3	788.23	249.56	123.47	759.17	119.82	124.20	35.61	283.28	1.29	80.92	619.08
4	787.73	249.54	123.46	757.45	119.81	124.22	37.11	284.78	1.23	106.00	594.00
5	787.21	249.53	123.44	755.66	119.81	124.23	38.68	286.36	1.18	130.18	569.82
6	786.67	249.50	123.43	753.78	119.80	124.25	40.33	288.01	1.14	153.51	546.49
7	786.10	249.48	123.41	751.80	119.79	124.27	42.06	289.74	1.09	176.01	523.99
8	785.51	249.46	123.39	749.73	119.78	124.29	43.87	291.55	1.04	197.70	502.30
9	784.88	249.44	123.37	747.56	119.77	124.31	45.78	293.46	1.00	218.63	481.37
10	784.22	249.41	123.35	745.28	119.76	124.33	47.78	295.46	0.96	238.80	461.20
11	783.53	249.39	123.33	742.89	119.75	124.35	49.88	297.56	0.92	258.27	441.73
12	782.81	249.36	123.31	740.37	119.74	124.37	52.09	299.77	0.88	277.04	422.96
13	782.04	249.33	123.29	737.73	119.73	124.40	54.41	302.10	0.84	295.14	404.86
14	781.24	249.30	123.26	734.94	119.72	124.42	56.86	304.54	0.81	312.60	387.40
							610.45	4078.94			

Propellant Load: 306 lbm

$I_{sp} = 225$

AERONOMY/"700"

TABLE 4-3A

PARKING ORBIT 230.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 700.LBM DRAG AREA 4.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 225. PULSING ISP 150. SPACECRAFT RADIUS 2.00FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	786.67	229.41	104.47	759.96	119.84	105.30	34.94	244.70	1.40	24.63	675.37
2	786.18	229.39	104.45	758.50	119.83	105.31	36.22	245.98	1.35	48.51	651.49
3	785.68	229.37	104.43	756.98	119.82	105.32	37.55	247.30	1.30	71.66	628.34
4	785.15	229.34	104.41	755.40	119.82	105.33	38.94	248.68	1.26	94.10	605.90
5	784.60	229.32	104.39	753.74	119.81	105.35	40.39	250.12	1.21	115.85	584.15
6	784.03	229.29	104.37	752.02	119.80	105.36	41.90	251.63	1.17	136.95	563.05
7	783.43	229.27	104.35	750.23	119.80	105.38	43.47	253.20	1.13	157.39	542.61
8	782.80	229.24	104.32	748.35	119.79	105.39	45.12	254.83	1.09	177.22	522.78
9	782.15	229.21	104.30	746.39	119.78	105.41	46.84	256.54	1.05	196.43	503.57
10	781.47	229.18	104.27	744.35	119.77	105.42	48.63	258.33	1.01	215.07	484.93
11	780.76	229.15	104.25	742.21	119.76	105.44	50.51	260.20	0.97	233.13	466.87
12	780.01	229.12	104.22	739.97	119.75	105.46	52.48	262.15	0.93	250.64	449.36
13	779.23	229.08	104.19	737.63	119.74	105.48	54.53	264.20	0.90	267.61	432.39
14	778.42	229.05	104.16	735.18	119.73	105.50	56.69	266.34	0.86	284.07	415.93
15	777.57	229.01	104.13	732.62	119.72	105.52	58.94	268.59	0.83	300.03	399.97
16	776.67	228.97	104.09	729.93	119.71	105.54	61.31	270.94	0.80	315.50	384.50

Propellant Load: 306 lbm
$I_{sp} = 225$

AERONOMY/"700"

TABLE 4-3B

PARKING ORBIT 200.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 700.LBM DRAG AREA 4.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 225. PULSING ISP 150. SPACECRAFT RADIUS 2.00FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	772.10	199.13	75.81	745.50	119.84	76.88	47.67	200.36	1.41	20.48	679.52
2	771.26	199.11	75.79	743.85	119.83	76.89	49.11	201.79	1.37	40.50	659.50
3	770.39	199.08	75.77	742.15	119.83	76.90	50.61	203.28	1.33	60.05	639.95
4	769.48	199.05	75.74	740.38	119.82	76.91	52.17	204.82	1.29	79.16	620.84
5	768.54	199.02	75.72	738.55	119.82	76.92	53.78	206.42	1.25	97.83	602.17
6	767.57	198.99	75.69	736.64	119.81	76.93	55.46	208.08	1.21	116.08	583.92
7	766.55	198.96	75.67	734.66	119.80	76.94	57.20	209.81	1.17	133.90	566.10
8	765.50	198.93	75.64	732.61	119.80	76.96	59.01	211.61	1.14	151.32	548.68
9	764.41	198.89	75.61	730.47	119.79	76.97	60.90	213.48	1.10	168.34	531.66
10	763.27	198.86	75.58	728.24	119.78	76.98	62.86	215.42	1.07	184.97	515.03
11	762.08	198.82	75.55	725.92	119.78	77.00	64.90	217.45	1.04	201.22	498.78
12	760.84	198.78	75.51	723.51	119.77	77.01	67.03	219.56	1.00	217.09	482.91
13	759.56	198.74	75.48	721.00	119.76	77.02	69.25	221.76	0.97	232.61	467.39
14	758.21	198.70	75.44	718.38	119.75	77.04	71.57	224.05	0.94	247.76	452.24
15	756.81	198.66	75.41	715.64	119.75	77.06	73.98	226.45	0.91	262.58	437.42
16	755.35	198.61	75.37	712.78	119.74	77.07	76.51	228.95	0.88	277.05	422.95
17	753.82	198.56	75.33	709.80	119.73	77.09	79.15	231.56	0.85	291.19	408.81
18	752.23	198.51	75.29	706.68	119.72	77.11	81.91	234.30	0.82	305.01	394.99
19	750.56	198.46	75.24	703.42	119.71	77.13	84.80	237.17	0.79	318.51	381.49
							1217.87	4116.32			

Propellant Load: 306 lbm

$I_{sp} = 225$

AERONOMY/"700"

TABLE 4-3C

PARKING ORBIT 250.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 700.LBM DRAG AREA 4.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300. PULSING ISP 150. SPACECRAFT RADIUS 2.00FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	789.15	249.60	123.50	762.49	119.84	124.17	32.71	280.38	1.40	21.41	678.59
2	788.81	249.58	123.49	761.30	119.83	124.18	33.74	281.41	1.36	42.24	657.76
3	788.45	249.57	123.48	760.08	119.83	124.19	34.81	282.49	1.32	62.50	637.50
4	788.09	249.56	123.47	758.81	119.82	124.20	35.92	283.60	1.28	82.21	617.79
5	787.71	249.54	123.46	757.49	119.81	124.22	37.08	284.75	1.24	101.38	598.62
6	787.31	249.53	123.45	756.13	119.81	124.23	38.27	285.94	1.20	120.03	579.97
7	786.90	249.51	123.43	754.72	119.80	124.24	39.50	287.18	1.16	138.16	561.84
8	786.48	249.50	123.42	753.26	119.80	124.25	40.78	288.46	1.13	155.81	544.19
9	786.04	249.48	123.41	751.75	119.79	124.27	42.11	289.79	1.09	172.97	527.03
10	785.59	249.46	123.39	750.17	119.78	124.28	43.49	291.16	1.06	189.66	510.34
11	785.12	249.45	123.38	748.55	119.77	124.30	44.92	292.59	1.02	205.90	494.10
12	784.63	249.43	123.37	746.85	119.77	124.31	46.40	294.08	0.99	221.69	478.31
13	784.12	249.41	123.35	745.10	119.76	124.33	47.94	295.62	0.96	237.05	462.95
14	783.59	249.39	123.33	743.28	119.75	124.35	49.54	297.22	0.93	252.00	448.00
15	783.05	249.37	123.32	741.39	119.74	124.36	51.20	298.88	0.90	266.53	433.47
16	782.48	249.35	123.30	739.42	119.74	124.38	52.93	300.61	0.87	280.67	419.33
17	781.89	249.33	123.28	737.38	119.73	124.40	54.72	302.40	0.84	294.43	405.57
18	781.27	249.30	123.26	735.25	119.72	124.42	56.59	304.27	0.81	307.80	392.20
							782.65	5240.83			

Propellant Load: 306 lbm

$I_{sp} = 300$

AERONOMY/"770"

TABLE 4-3D

PARKING ORBIT 230.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 700.LBM DRAG AREA 4.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300. PULSING ISP 150. SPACECRAFT RADIUS 2.00FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	786.67	229.41	104.47	760.06	119.84	105.30	34.85	244.62	1.41	18.90	681.10
2	786.30	229.39	104.46	758.95	119.83	105.30	35.82	245.58	1.37	37.35	662.65
3	785.92	229.38	104.44	757.81	119.83	105.31	36.83	246.58	1.33	55.37	644.63
4	785.52	229.36	104.43	756.63	119.82	105.32	37.86	247.61	1.30	72.97	627.03
5	785.12	229.34	104.41	755.41	119.82	105.33	38.93	248.67	1.26	90.15	609.85
6	784.70	229.32	104.40	754.15	119.81	105.34	40.03	249.77	1.23	106.93	593.07
7	784.27	229.30	104.38	752.86	119.81	105.35	41.16	250.90	1.19	123.31	576.69
8	783.82	229.29	104.36	751.52	119.80	105.36	42.34	252.07	1.16	139.31	560.69
9	783.36	229.26	104.34	750.13	119.80	105.38	43.55	253.27	1.13	154.94	545.06
10	782.88	229.24	104.33	748.71	119.79	105.39	44.81	254.52	1.10	170.20	529.80
11	782.39	229.22	104.31	747.23	119.78	105.40	46.10	255.81	1.07	185.10	514.90
12	781.88	229.20	104.29	745.70	119.78	105.41	47.44	257.14	1.04	199.64	500.36
13	781.35	229.18	104.27	744.12	119.77	105.42	48.83	258.52	1.01	213.85	486.15
14	780.81	229.15	104.25	742.49	119.76	105.44	50.26	259.95	0.98	227.73	472.27
15	780.24	229.13	104.23	740.80	119.76	105.45	51.75	261.43	0.95	241.27	458.73
16	779.66	229.10	104.21	739.05	119.75	105.47	53.28	262.96	0.92	254.50	445.50
17	779.05	229.07	104.18	737.24	119.74	105.48	54.88	264.54	0.90	267.42	432.58
18	778.43	229.05	104.16	735.37	119.74	105.50	56.52	266.18	0.87	280.04	419.96
19	777.78	229.02	104.13	733.43	119.73	105.51	58.23	267.88	0.84	292.36	407.64
20	777.11	228.99	104.11	731.41	119.72	105.53	60.00	269.64	0.82	304.39	395.61
21	776.41	228.96	104.08	729.33	119.71	105.54	61.84	271.47	0.80	316.14	383.86

Propellant Load: 306 lbm

$I_{sp} = 300$

AERONOMY/"700"

TABLE 4-3E

PARKING ORBIT 200.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 700.LBM DRAG AREA 4.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300. PULSING ISP 150. SPACECRAFT RADIUS 2.00FT

1959 ARDC ATMOSPHERE SPIN RATE 10.0RPM

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	772.10	199.13	75.81	745.57	119.84	76.88	47.61	200.30	1.41	15.76	684.24
2	771.46	199.11	75.80	744.31	119.83	76.89	48.71	201.39	1.38	31.25	668.75
3	770.80	199.09	75.78	743.02	119.83	76.89	49.84	202.52	1.35	46.46	653.54
4	770.12	199.07	75.76	741.70	119.83	76.90	51.01	203.67	1.32	61.40	638.60
5	769.42	199.05	75.74	740.33	119.82	76.91	52.21	204.86	1.29	76.08	623.92
6	768.70	199.03	75.72	738.93	119.82	76.92	53.45	206.09	1.26	90.50	609.50
7	767.96	199.00	75.70	737.49	119.81	76.93	54.72	207.35	1.23	104.66	595.34
8	767.19	198.98	75.68	736.00	119.81	76.93	56.03	208.64	1.20	118.57	581.43
9	766.41	198.95	75.66	734.47	119.80	76.94	57.37	209.98	1.17	132.23	567.77
10	765.60	198.93	75.64	732.89	119.80	76.95	58.76	211.36	1.15	145.65	554.35
11	764.77	198.90	75.62	731.27	119.79	76.96	60.19	212.78	1.12	158.84	541.16
12	763.91	198.88	75.60	729.59	119.79	76.97	61.67	214.24	1.09	171.79	528.21
13	763.03	198.85	75.57	727.86	119.78	76.98	63.19	215.75	1.07	184.51	515.49
14	762.11	198.82	75.55	726.08	119.78	76.99	64.76	217.30	1.04	197.01	502.99
15	761.17	198.79	75.52	724.25	119.77	77.00	66.38	218.91	1.02	209.28	490.72
16	760.20	198.76	75.50	722.35	119.77	77.02	68.05	220.57	0.99	221.34	478.66
17	759.20	198.73	75.47	720.40	119.76	77.03	69.78	222.28	0.97	233.18	466.82
18	758.16	198.70	75.44	718.38	119.76	77.04	71.56	224.05	0.94	244.81	455.19
19	757.09	198.67	75.41	716.29	119.75	77.05	73.41	225.87	0.92	256.24	443.76
20	755.99	198.63	75.39	714.14	119.74	77.06	75.31	227.76	0.90	267.47	432.53
21	754.85	198.60	75.36	711.91	119.74	77.08	77.28	229.71	0.87	278.49	421.51
22	753.67	198.56	75.32	709.61	119.73	77.09	79.32	231.73	0.85	289.32	410.68
23	752.44	198.52	75.29	707.22	119.72	77.11	81.43	233.82	0.83	299.96	400.04
24	751.18	[illegible]	[illegible]	704.75	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	389.58
							1525.65	5186.92			

AERONOMY/"700"
TABLE 4-3F

Propellant Load: 306 lbm
$I_{sp} = 300$

PARKING ORBIT 250.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 5.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300.

1959 ARDC ATMOSPHERE

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	792.09	249.71	123.59	772.66	119.88	124.08	23.82	271.48	0.00	33.25	1166.75
2	791.86	249.70	123.58	771.88	119.88	124.08	24.50	272.16	0.00	65.67	1134.33
3	791.63	249.69	123.58	771.08	119.87	124.09	25.20	272.86	0.00	97.26	1102.74
4	791.39	249.68	123.57	770.25	119.87	124.10	25.92	273.59	0.00	128.05	1071.95
5	791.14	249.67	123.56	769.39	119.87	124.11	26.67	274.34	0.00	158.07	1041.93
6	790.89	249.66	123.55	768.51	119.86	124.11	27.44	275.11	0.00	187.32	1012.68
7	790.62	249.65	123.55	767.60	119.86	124.12	28.23	275.90	0.00	215.84	984.16
8	790.35	249.64	123.54	766.66	119.85	124.13	29.05	276.72	0.00	243.63	956.37
9	790.07	249.63	123.53	765.69	119.85	124.14	29.90	277.57	0.00	270.72	929.28
10	789.78	249.62	123.52	764.69	119.85	124.15	30.77	278.44	0.00	297.12	902.88
11	789.48	249.61	123.51	763.66	119.84	124.16	31.68	279.35	0.00	322.86	877.14
12	789.18	249.60	123.50	762.60	119.84	124.17	32.61	280.28	0.00	347.94	852.06
13	788.86	249.59	123.49	761.49	119.83	124.18	33.57	281.24	0.00	372.39	827.61
14	788.53	249.57	123.48	760.36	119.83	124.19	34.57	282.24	0.00	396.22	803.78
15	788.19	249.56	123.47	759.18	119.82	124.20	35.60	283.27	0.00	419.45	780.55
16	787.84	249.55	123.46	757.97	119.82	124.21	36.66	284.33	0.00	442.09	757.91
17	787.47	249.53	123.45	756.71	119.81	124.22	37.76	285.43	0.00	464.16	735.84

Propellant Load: 645 lbm

$I_{sp} = 300$

AERONOMY/"550"T

TABLE 4-4A

18	787.10	249.52	123.44	755.41	119.81	124.23	38.90	286.57	0.00	485.67	714.33
19	786.71	249.51	123.43	754.07	119.80	124.25	40.07	287.75	0.00	506.63	693.37
20	786.31	249.49	123.42	752.68	119.79	124.26	41.29	288.96	0.00	527.07	672.93
21	785.89	249.48	123.40	751.25	119.79	124.27	42.55	290.22	0.00	546.98	653.02
22	785.46	249.46	123.39	749.76	119.78	124.29	43.85	291.53	0.00	566.40	633.60
23	785.02	249.44	123.38	748.22	119.77	124.30	45.20	292.88	0.00	585.32	614.68
24	784.55	249.43	123.36	746.63	119.77	124.31	46.60	294.28	0.00	603.76	596.24
25	784.08	249.41	123.35	744.97	119.76	124.33	48.05	295.73	0.00	621.74	578.26
26	783.58	249.39	123.33	743.24	119.75	124.35	49.55	297.23	0.00	639.26	560.74
27	783.07	249.37	123.32	741.49	119.74	124.36	51.11	298.79	0.00	656.34	543.66

TABLE 4-4A (Continued)

PARKING ORBIT 230.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 5.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300.

1959 ARDC ATMOSPHERE

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	790.28	229.57	104.61	770.89	119.88	105.21	25.38	235.19	0.00	28.86	1171.14
2	790.04	229.56	104.60	770.17	119.88	105.21	26.01	235.81	0.00	57.11	1142.89
3	789.79	229.55	104.59	769.43	119.87	105.22	26.65	236.46	0.00	84.74	1115.26
4	789.54	229.54	104.58	768.68	119.87	105.23	27.31	237.11	0.00	111.78	1088.22
5	789.28	229.53	104.57	767.90	119.87	105.23	27.99	237.79	0.00	138.24	1061.76
6	789.01	229.51	104.56	767.10	119.87	105.24	28.69	238.49	0.00	164.14	1035.86
7	788.74	229.50	104.55	766.28	119.86	105.24	29.41	239.20	0.00	189.47	1010.53
8	788.46	229.49	104.54	765.43	119.86	105.25	30.15	239.94	0.00	214.26	985.74
9	788.17	229.48	104.53	764.56	119.85	105.26	30.91	240.70	0.00	238.52	961.48
10	787.87	229.46	104.51	763.67	119.85	105.27	31.69	241.47	0.00	262.26	937.74
11	787.56	229.45	104.50	762.75	119.85	105.27	32.50	242.28	0.00	285.48	914.52
12	787.25	229.44	104.49	761.80	119.84	105.28	33.33	243.10	0.00	308.21	891.79
13	786.92	229.42	104.48	760.83	119.84	105.29	34.18	243.95	0.00	330.45	869.55
14	786.59	229.41	104.47	759.83	119.84	105.30	35.06	244.82	0.00	352.21	847.79
15	786.24	229.39	104.45	758.80	119.83	105.31	35.96	245.72	0.00	373.50	826.50
16	785.89	229.38	104.44	757.73	119.83	105.31	36.89	246.64	0.00	394.34	805.66
17	785.52	229.36	104.43	756.64	119.82	105.32	37.85	247.60	0.00	414.73	785.27

Propellant Load: 645 lbm

$I_{sp} = 300$

AERONOMY/"550"T

TABLE 4-4B

18	785.15	229.34	104.41	755.52	119.82	105.33	38.83	248.58	0.00	434.68	765.32
19	784.76	229.33	104.40	754.36	119.81	105.34	39.85	249.59	0.00	454.20	745.80
20	784.36	229.31	104.38	753.16	119.81	105.35	40.90	250.63	0.00	473.30	726.70
21	783.95	229.29	104.37	751.93	119.80	105.36	41.98	251.71	0.00	491.99	708.01
22	783.53	229.27	104.35	750.66	119.80	105.37	43.09	252.81	0.00	510.28	689.72
23	783.09	229.25	104.33	749.35	119.79	105.38	44.24	253.95	0.00	528.17	671.83
24	782.64	229.23	104.32	748.00	119.79	105.39	45.42	255.13	0.00	545.69	654.31
25	782.17	229.21	104.30	746.61	119.78	105.40	46.64	256.35	0.00	562.82	637.18
26	781.69	229.19	104.28	745.18	119.78	105.42	47.90	257.60	0.00	579.59	620.41
27	781.20	229.17	104.26	743.69	119.77	105.43	49.20	258.90	0.00	595.99	604.01
28	780.69	229.15	104.24	742.17	119.76	105.44	50.55	260.23	0.00	612.05	587.95
29	780.16	229.12	104.22	740.59	119.76	105.45	51.94	261.61	0.00	627.76	572.24
30	779.62	229.10	104.20	738.96	119.75	105.47	53.37	263.04	0.00	643.13	556.87
31	779.05	229.07	104.18	737.27	119.74	105.48	54.85	264.51	0.00	658.17	541.83

TABLE 4-4B (Continued)

PARKING ORBIT 200.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 1200.LBM DRAG AREA 5.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300.

1959 ARDC ATMOSPHERE

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	779.66	199.37	76.01	760.32	119.88	76.79	34.66	187.46	0.00	23.06	1176.94
2	779.26	199.35	76.00	759.54	119.88	76.79	35.34	188.14	0.00	45.76	1154.24
3	778.85	199.34	75.99	758.75	119.88	76.80	36.04	188.83	0.00	68.11	1131.89
4	778.43	199.33	75.98	757.93	119.87	76.80	36.75	189.54	0.00	90.10	1109.90
5	778.00	199.32	75.97	757.10	119.87	76.81	37.48	190.26	0.00	111.75	1088.25
6	777.57	199.30	75.96	756.25	119.87	76.81	38.23	191.00	0.00	133.05	1066.95
7	777.12	199.29	75.95	755.37	119.87	76.82	39.00	191.76	0.00	154.02	1045.98
8	776.66	199.27	75.94	754.48	119.86	76.82	39.78	192.54	0.00	174.66	1025.34
9	776.19	199.26	75.92	753.56	119.86	76.83	40.59	193.34	0.00	194.98	1005.02
10	775.71	199.24	75.91	752.62	119.86	76.84	41.41	194.16	0.00	214.98	985.02
11	775.22	199.23	75.90	751.66	119.86	76.84	42.25	194.99	0.00	234.66	965.34
12	774.71	199.21	75.88	750.68	119.85	76.85	43.12	195.85	0.00	254.04	945.96
13	774.19	199.20	75.87	749.67	119.85	76.85	44.01	196.73	0.00	273.11	926.89
14	773.66	199.18	75.86	748.63	119.85	76.86	44.92	197.63	0.00	291.88	908.12
15	773.12	199.16	75.84	747.57	119.84	76.87	45.85	198.56	0.00	310.35	889.65
16	772.56	199.15	75.83	746.48	119.84	76.87	46.81	199.50	0.00	328.54	871.46
17	771.99	199.13	75.81	745.36	119.84	76.88	47.79	200.48	0.00	346.44	853.56
18	771.40	199.11	75.80	744.22	119.83	76.89	48.79	201.47	0.00	364.06	835.94
19	770.80	199.09	75.78	743.04	119.83	76.89	49.83	202.50	0.00	381.40	818.60
20	770.18	199.07	75.76	741.84	119.83	76.90	50.89	203.55	0.00	398.47	801.53

Propellant Load: 645 lbm

$I_{sp} = 300$

AERONOMY/"550"T

TABLE 4-4C

21	769.54	199.05	75.75	740.60	119.82	76.91	51.98	204.63	0.00	415.27	784.73
22	768.89	199.03	75.73	739.33	119.82	76.92	53.10	205.74	0.00	431.80	768.20
23	768.22	199.01	75.71	738.02	119.81	76.92	54.25	206.88	0.00	448.08	751.92
24	767.53	198.99	75.69	736.68	119.81	76.93	55.43	208.05	0.00	464.10	735.90
25	766.83	198.97	75.67	735.30	119.81	76.94	56.64	209.25	0.00	479.87	720.13
26	766.10	198.95	75.65	733.88	119.80	76.95	57.89	210.49	0.00	495.39	704.61
27	765.35	198.92	75.63	732.43	119.80	76.96	59.17	211.76	0.00	510.67	689.33
28	764.58	198.90	75.61	730.93	119.79	76.96	60.49	213.07	0.00	525.71	674.29
29	763.79	198.87	75.59	729.39	119.79	76.97	61.85	214.42	0.00	540.51	659.49
30	762.98	198.85	75.57	727.80	119.78	76.98	63.25	215.80	0.00	555.08	644.92
31	762.15	198.82	75.55	726.17	119.78	76.99	64.68	217.23	0.00	569.42	630.58
32	761.29	198.80	75.53	724.49	119.77	77.00	66.16	218.69	0.00	583.54	616.46
33	760.40	198.77	75.50	722.77	119.77	77.01	67.69	220.21	0.00	597.43	602.57
34	759.49	198.74	75.48	720.98	119.76	77.02	69.26	221.76	0.00	611.11	588.89
35	758.54	198.71	75.45	719.15	119.76	77.04	70.88	223.37	0.00	624.57	575.43
36	757.58	198.68	75.43	717.26	119.75	77.05	72.55	225.03	0.00	637.82	562.18
37	756.58	198.65	75.40	715.31	119.75	77.06	74.27	226.73	0.00	650.86	549.14

TABLE 4-4C (Continued)

PARKING ORBIT 250.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 380.LBM DRAG AREA 4.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300.

1959 ARDC ATMOSPHERE

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	780.01	249.26	123.23	730.93	119.70	124.46	60.39	308.07	0.00	11.93	368.07
2	779.37	249.23	123.21	728.69	119.69	124.48	62.36	310.04	0.00	23.55	356.45
3	778.69	249.21	123.19	726.37	119.68	124.50	64.41	312.09	0.00	34.89	345.11
4	777.99	249.18	123.17	723.95	119.67	124.52	66.53	314.22	0.00	45.93	334.07

Propellant Load: 45.9 lbm

$I_{sp} = 300$

AERONOMY/"550"S

TABLE 4-5A

PARKING ORBIT 230.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 380.LBM DRAG AREA 4.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300.

1959 ARDC ATMOSPHERE

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	775.44	228.91	104.05	726.46	119.70	105.57	64.37	273.98	0.00	10.63	369.37
2	774.74	228.88	104.02	724.34	119.69	105.59	66.23	275.84	0.00	21.02	358.98
3	774.01	228.85	103.99	722.15	119.68	105.60	68.16	277.76	0.00	31.20	348.80
4	773.25	228.82	103.96	719.88	119.67	105.62	70.17	279.75	0.00	41.16	338.84
5	772.46	228.78	103.93	717.53	119.66	105.64	72.24	281.82	0.00	50.90	329.10

Propellant Load: 45.9 lbm

$I_{sp} = 300$

AERONOMY/"550"S

TABLE 4-5B

PARKING ORBIT 200.KM PERIGEE 800.KM APOGEE LOW SWEEP PERIGEE 120.KM

SWEEPS PER MANEUVER 15 DAYS BETWEEN MANEUVERS 15

LIFT OFF WEIGHT 380.LBM DRAG AREA 4.00SQ.FT. DRAG COEF. 2.6

PROPELLANT ISP 300.

1959 ARDC ATMOSPHERE

MNVR	HA1	HP1	DVA1	HA2	HP2	DVA2	DVP2	TDV	WPACS	WPT	W
1	748.60	198.40	75.19	699.77	119.70	77.15	88.03	240.37	0.00	9.34	370.66
2	747.31	198.36	75.15	697.24	119.69	77.17	90.27	242.59	0.00	18.53	361.47
3	745.97	198.32	75.12	694.63	119.68	77.18	92.59	244.89	0.00	27.58	352.42
4	744.58	198.28	75.08	691.92	119.68	77.20	94.99	247.26	0.00	36.49	343.51
5	743.15	198.23	75.04	689.12	119.67	77.21	97.47	249.73	0.00	45.25	334.75

Propellant Load: 45.9 lbm

$I_{sp} = 300$

AERONOMY/"550"S
TABLE 4-5C

Vehicle Design	Type of Propellant *	Parking Orbit Perigee **		
		250 km	230 km	200 km
		Number of Basic Maneuvers ***		
Aeronomy/"1200" Target	Monopropellant	17	19	23
	Bipropellant	22	25	29
Aeronomy/"1200" Optimum	Monopropellant	20	23	27
	Bipropellant	27	30	35
Aeronomy/"700"	Monopropellant	14	16	19
	Bipropellant	18	21	24
Aeronomy/"550"T	Bipropellant	27	31	37
Aeronomy/"550"S	Bipropellant	4	4	5

* Monopropellant $I_{sp} = 225$

Bipropellant $I_{sp} = 300$

** Parking Orbit Apogee = 800 km

*** Basic Maneuver Defined in Section 4.1.1

SUMMARY OF MISSION TABLES

TABLE 4-6

Table 4-6 emphasis the gains in mission life that may be attained by lowering the perigee of the parking orbit. This is due to the fact that only about 15 percent (250 km parking orbit perigee) to 30 percent (200 km parking orbit perigee) of the total energy available is used to overcome the effects of aerodynamic drag. The major portion of the available energy is used to transfer between the parking orbit and the low sweep orbit. It is recommended that additional study be performed to determine what energy balance (i.e. ratio of a energy used in conteracting drag to energy used in changing orbits) results in the longest mission lifetime for a given vehicle propellant load.

The lowering of parking orbit perigee is shown by Table 4-6 to have a significant effect. A cursory look at the effects of changing parking orbit apogee revealed that raising the apogee results in increased total lifetime while lowering it decreases total lifetime. This is of course as would be expected, since the energy of the initial orbit is increased. The effects of raising apogee are not as pronounced as those due to lower perigee, however, it is indicated that this is an area for additional study.

4.2 OPERATION ANALYSIS

The nature of the propelled aeronomy satellite is such that the sequence of operations required to support the basic mission is relatively complicated and extensive. The sequence is summarized in Section 2.4 and in Figure 2-1. Further operational requirements are given in Section 6, Specifications, Paragraph 3.1.1. To serve brevity, only salient points are discussed here.

4.2.1 Launch Site Operation

The spacecraft shall, upon delivery at the launch site, be tested to insure correct operation of all systems. Special tests required at the launch site will be conducted utilizing test procedures prepared for this purpose. In particular, vehicle interfaces will be verified and spin-balance tests conducted. A special test fixture to perform the spin balance may be required to accomondate the 5 foot diameter of the satellite. The satellite may be loaded with propellant during part of the test.

Special handling procedures will be prepared for the satellite due to its propellant load. These procedures will include detailed propellant loading instructions and personnel safety requirements. Use of prime and subcontractors personnel is anticipated.

All other integrated launch vehicle-spacecraft compatability tests are standard and impose no outstanding problem areas.

4.2.2 Net Operation

The net operations are summarized in the following paragraphs for a maneuver.

The attitude information from the satellite is received and transmitted to a ground based computer. In parallel, the tracking network data transmits the orbital information to the ground based computer. The computer outputs the required attitude change (rotation of thrust/spin axis) to the appropriate Stadan station which transmits the information to the satellite. Attitude verification is obtained and returned to the computer.

The computer outputs the corrections in attitude, if any, and selects the station to transmit propulsion information and the commands required to restore cruise attitude. The cruise attitude is verified and corrections, if any, output by the computer.

The above sequence is repeated for each propulsive phase of the maneuver.

For example, once every 15 days apogee position is verified, maneuver time selected, a 90 degree turn completed and corrected within a few minutes, thrust phase completed and corrected in the next few minutes and another 90 degree turn completed and corrected. An orbit may elapse between initial and corrective thrusting, but the actual net time should not exceed two hours for the deorbit maneuver. The next operation takes place about 24 hours later, and consists of two periods as given, each requiring perhaps two hours of actual involvement including standby time.

The actual operational time of the Stadan Network is therefore not as extensive as inferred from the complexity of the maneuver; since all that is required of the network is that it perform as a link between the ground based computer and the satellite. Further study is required to set forth all necessary requirements.

-5-

PRELIMINARY DESIGN STUDIES

Sections 2 and 3 summarize and compare the two basic design concepts employed. This section presents details on which the parametric performance estimates are based. The Aeronomy/550 approach was developed first and is presented in the next section below; the Aeronomy/Spin Stabilized approach is presented beginning in Section 5.2.

The major comparisons between approaches have already been covered in the preceding sections. Only one additional factor is brought out here because it is peculiar to the data system, a system which did not markedly influence the design concepts. This factor is the relative orbital coverage of the two approaches. It is found that both systems require tape recorders, but the key requirement of high sampling rates (8640 bps) for the Aeronomy/SS to decouple spin effects does not apply to the Aeronomy/550. Both systems face the same playback time: the Aeronomy/SS obtains only fractional orbit coverage as a result. Table 5-1 illustrates Typical Aeronomy Experiments considered to obtain the above data requirements.

5.1 THE AERONOMY/550 APPROACH

Initial effort under this contract was spent in preliminary design adaption of the 550 Standard Payload Module concept to a low-orbit aeronomy mission. This propelled satellite concept had been studied under MSFC contract NAS8-20253, and the present study presents a specific adaption to the Scout launch vehicle. Toward the end of the study a comparative performance was also estimated for a 1200 lbm liftoff mass version, with characteristics noted in Table 3-6. The overall configuration is shown in figure 5-1, a block diagram in figure 5-2, and the packaging concept in figures 5-3, and 5-3a.

EXPERIMENT	BY	WEIGHT	POWER	MEASUREMENTS	APPROXIMATE Range-Torr & Comments
Double focusing Mass. Spectrometer C. S. C.	Spencer & Reber	$\sim$12 lbs. Best W/4 lbs. magnet	$\sim$15 -20 W	Total ion beam current and eight collectors sampled	10^{-4} to 10^{-10} Torr
Quadrupole Spectrometer Lockheed (Palo Alto)	D. McKibben T. Flowerday	$\sim$7 lbs. "main" instrument	7 watts tot. R.F. $\sim$ 3w other $\sim$ 4w	RF sample Elec. field sample 12 to 46 masses observed	Some drifting problems, low dynamic range.
Quadrupole Spectrometer Univ. of Michigan	E. J. Schaefer	$\sim$ 2-3 lbs. and electronics	3w Rf pwr only	14 to 52 masses observed	Resolution sacrificed for stability & higher sensitivity; down to 10^{-6} Torr.
Bennett R. F. Spectrometer Two Stage Single Cycle	H. Taylor-G. S. F. C. C. Smith	20 lbs.	$\sim$ 6.0 w	1-7 or 7-45 masses observed	Lower limits 10^{-10} Torr.
Cold Cathode inverted Magnetron Red-Head Type	in H. Vac. Installations	2-4 lb magnet $\sim$ 2 lbs elect. Total 6 lbs.	2-6 Kva Supply 1 watt	0-5 V analog	Time delay at low pressures linear output 10^{-10} torr less accuracy than thermonic type
Hot Cathode Magnetron Ionization Bayard-Alpert type	in Hi Vac. Installations	2-4 lb magnet $\sim$3 lbs. elect. Total $\sim$ 7 lbs.	+200-300 V -25 -50 V filament power needed 10-15 w total	Sensitive electrometer needed 0-5 V analog	Filament changed by gassive composition, fragile filament 10^{-12} torr.
Langmuir Probes	Exp. XVII and other Space Craft	.5 to 1 lb.	2.5 Electrometer .5 Sweep Gen. $\sim$ 1 watt total	0-5 V analog	Common experiment used for cross coorolation with other experiments
Ionization Guages	N.R.C.	Electrometer Elec. 1 lb.	.5 watt	0-5 V analog	Needs sensitive electrometer, needs source with attendent problems 10^{-7} Torr.

Ref: C.D. Schrader; Survey of Rocket- and Satellite-Borne mass Spectrometers, 1962
A. Guthrie, Vacuum Technology 1963

TYPICAL AERONOMY EXPERIMENTS

TABLE 5-1

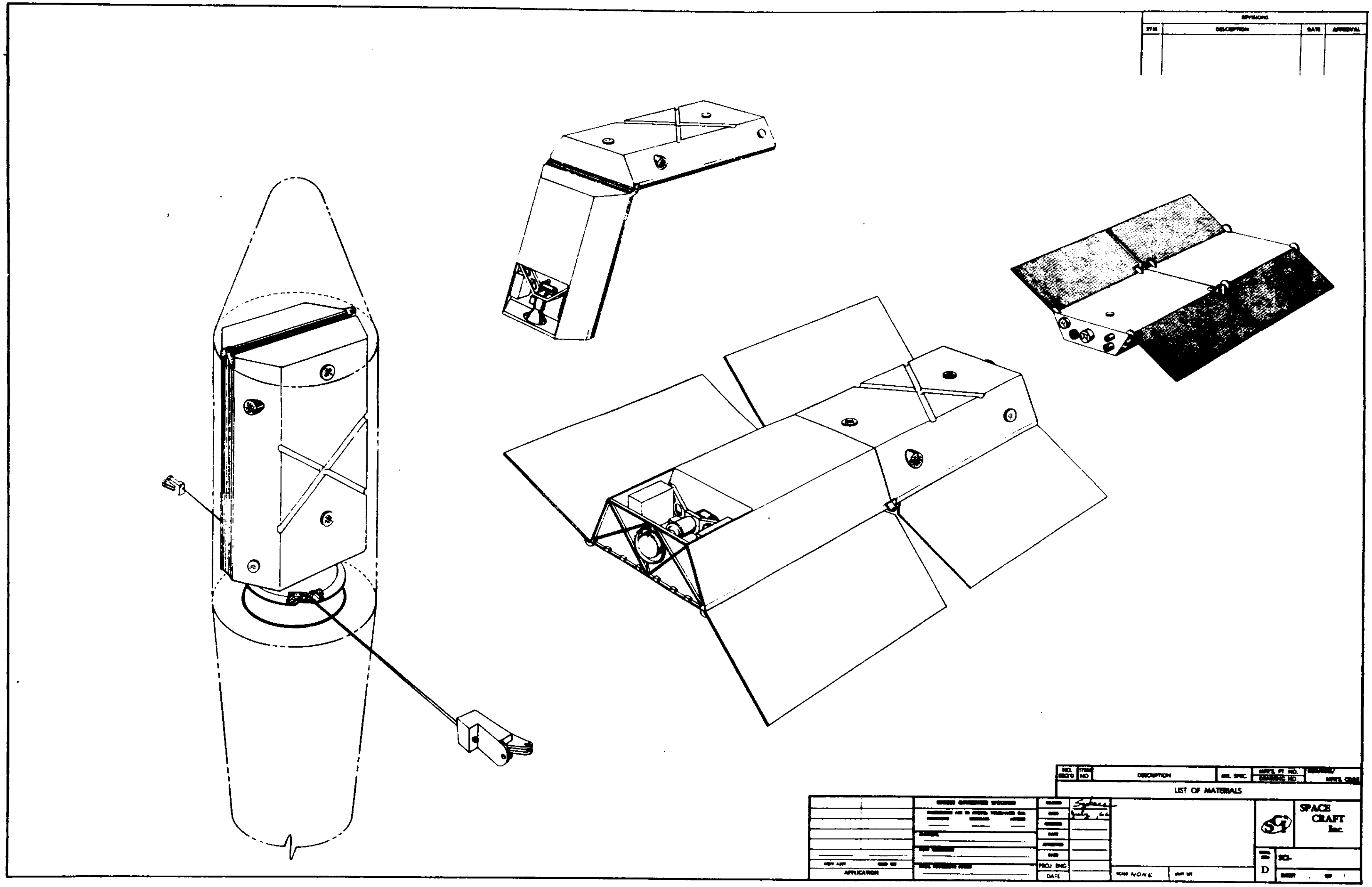
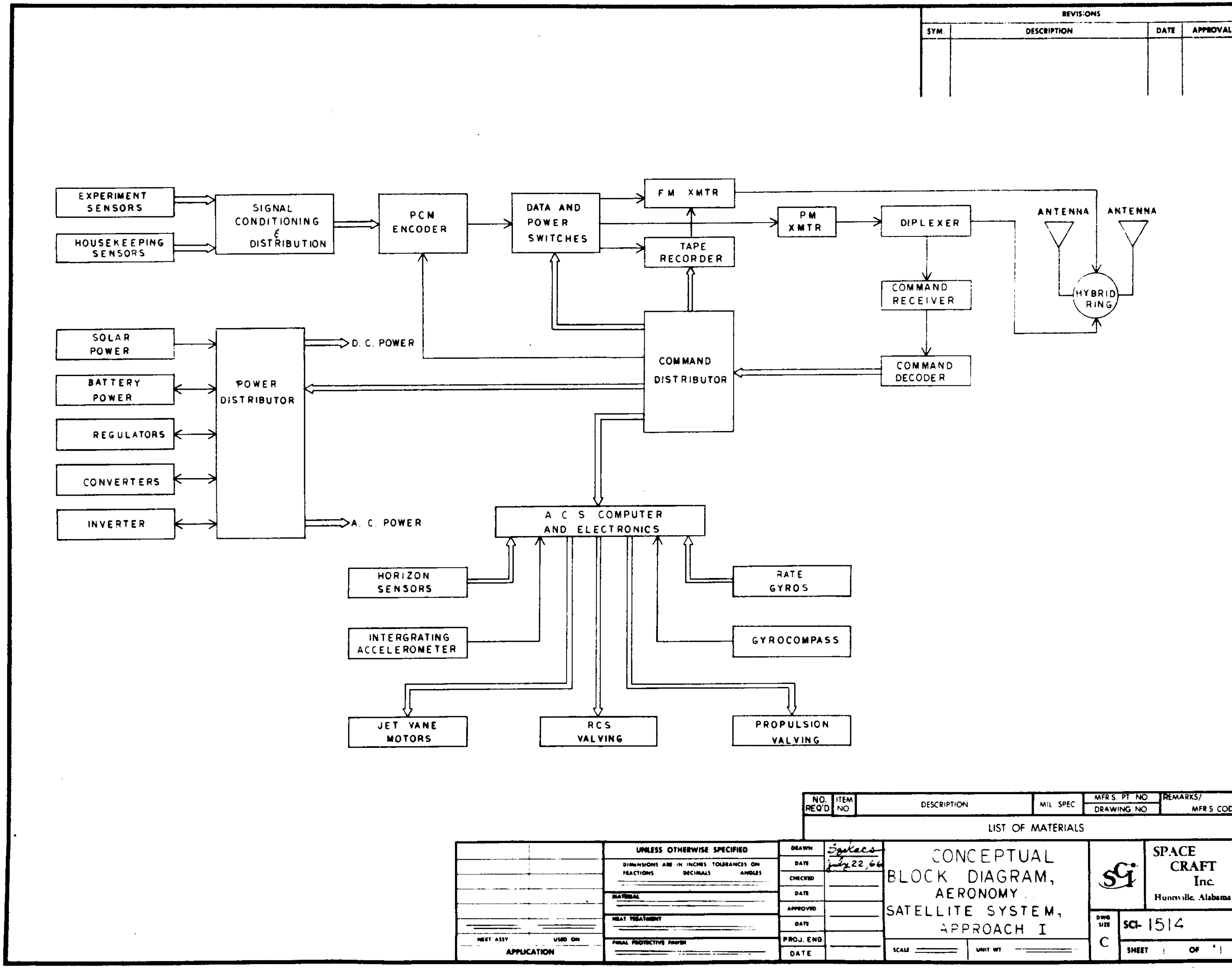
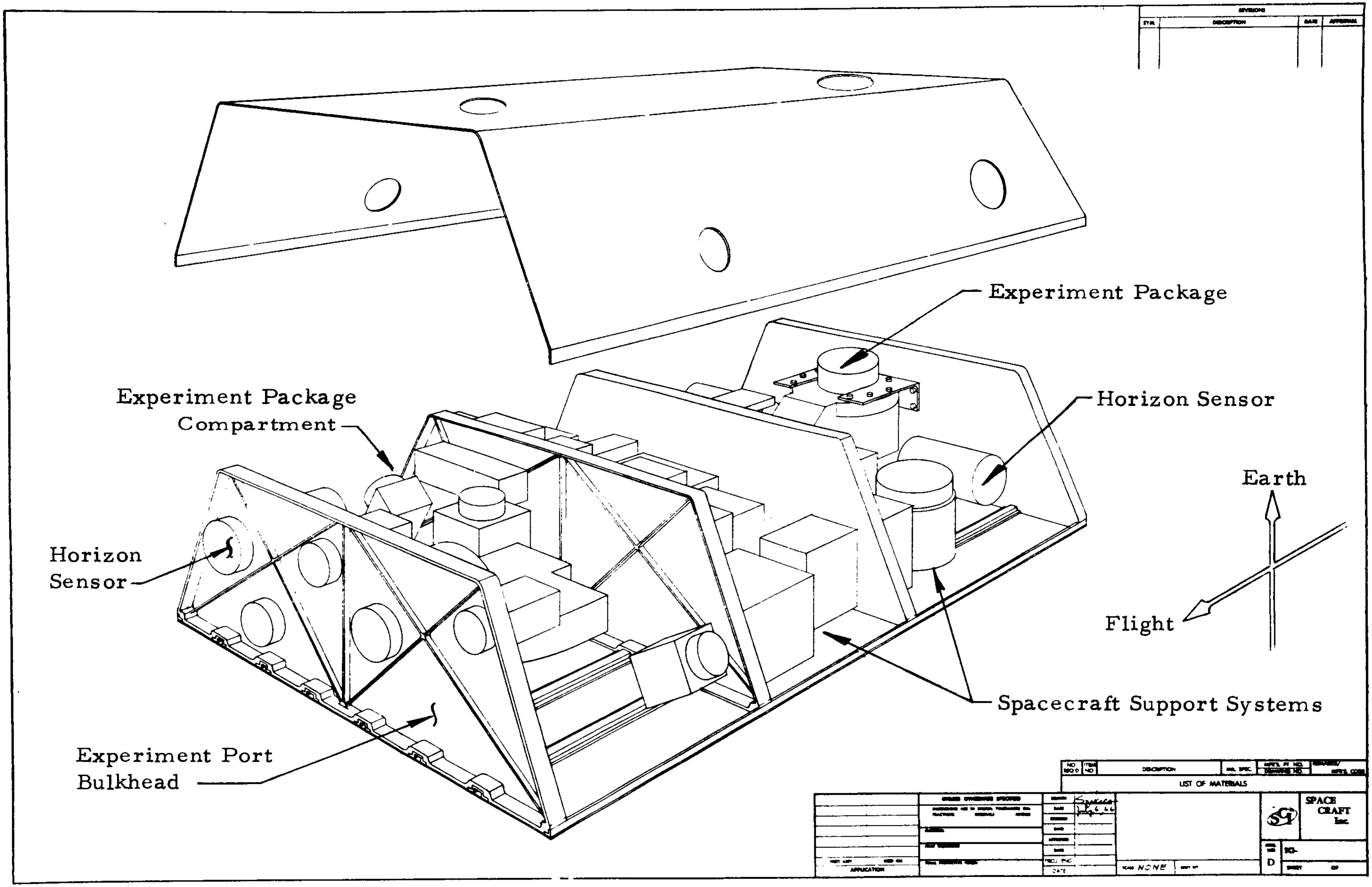


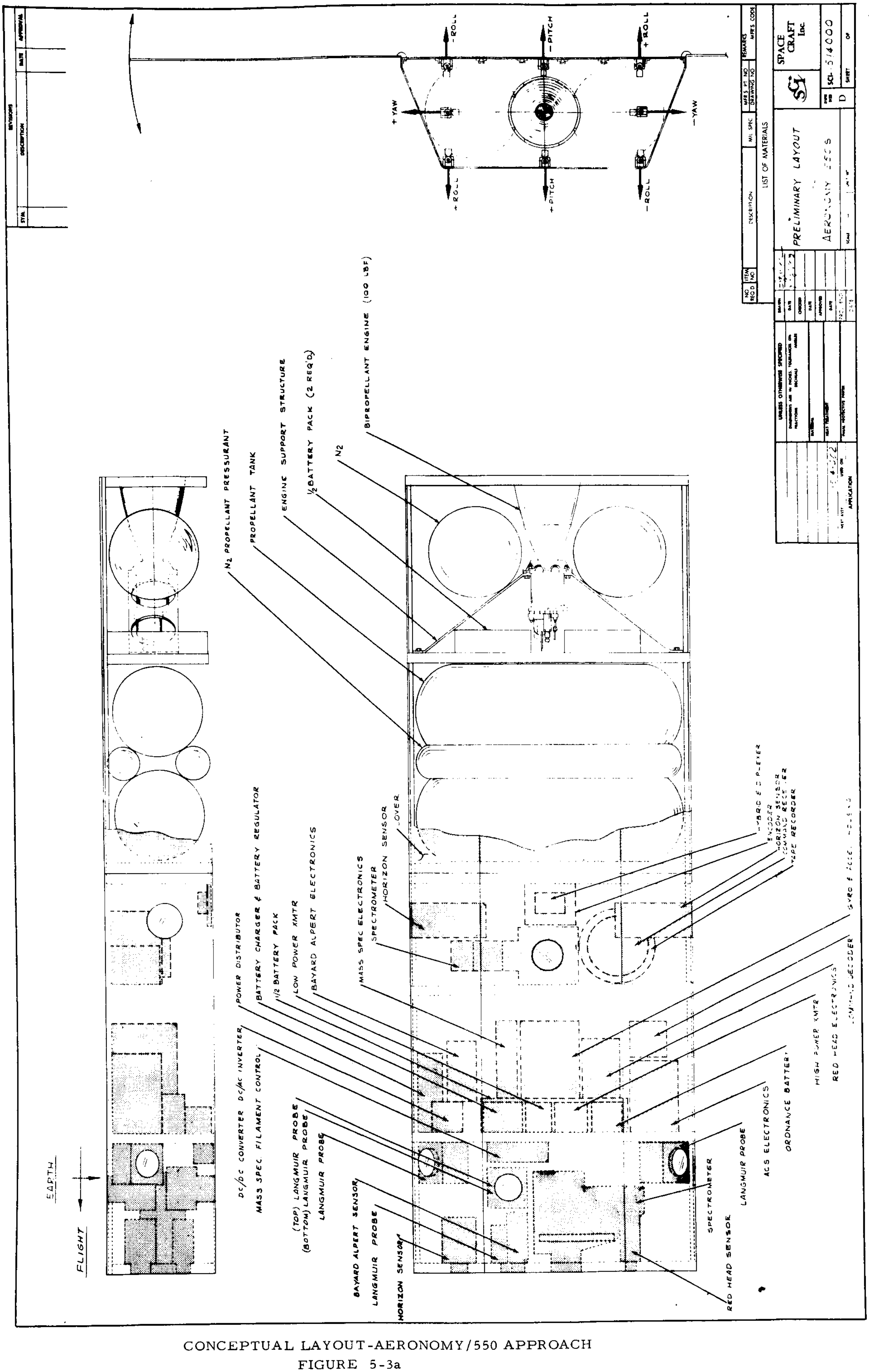
FIGURE 5-1
OVERALL CONFIGURATION, AERONOMY/550S SATELLITE



BLOCK DIAGRAM, AERONOMY/550S SYSTEM



PACKAGING CONCEPT AERONOMY/550S
FIGURE 5-3



CONCEPTUAL LAYOUT-AERONOMY/550 APPROACH
FIGURE 5-3a

5.1.1 Configuration Analysis

Two concepts based on the Space Craft, Inc. "550 SPM" design were investigated. Basic systems are identical, but the Aeronomy/550S folds into the Scout envelope, while the Aeronomy/550T has a rigid body with an extended propulsion section to take advantage of the higher Thor-Delta lift capabilities.

Aeronomy/550S

Scout payload capabilities were obtained. A '550' SPM satellite configuration was adapted to the Scout. Preliminary analysis indicated that the standard payload envelope (33" cylindrical section 30" in diameter plus cone frustrum) would accept a modified satellite configuration which exhibited the desired nominal one-month life. However, the Scout payload envelope was marginal with respect to available volume and as a consequence, both experiment volume allocation and solar panel area were undersize. An effort to reduce both spacecraft component volume and power was therefore initiated.

Alternate Scout payload envelopes were investigated. Ling-Tempco Vought representatives provided information on a new standard payload envelope having a 48" nominal cylindrical section, plus frustrum section. These values made possible a substantial increase in solar cell area and experiment volume for the trial configuration. Preliminary detail calculations indicate that approximately 60 watts of continuous power are available at the battery taps. The calculations include factors for cell derating, worst orbit inclination for eastward Wallops Island launch, and worst solar incidence angle. Net continuous power to the experimenter is about 40-44 watts.

Spacecraft power and volume reductions (relative to the 550 baseline design) were also carried out. The altitude restriction to 100 km (100 NM) nominal maximum permits radiated power reduction; choice of a smaller PCM encoder also permits power reduction; these major adjustments plus less significant changes reduce mass within the Scout 400 lbm nominal lift capability to 100 NM circular orbit injection. The design can carry approximately 50 lbm of prime experiment. This figure assumes that the structure including solar panel substructure, mechanisms, and satellite body sections weigh approximately 70 lbm and the battery approximately 50 lbm.

Aeronomy/550T

The Nimbus shroud envelope has a minimum of 100 inch length available at 5 feet diameter. This permits an Aeronomy/550T designed for 1200 lbm initial mass, nominal. The design is included for completeness of the relative comparisons: it is pointed out that the resulting mass fractions are very similar to those of the Aeronomy/SS 1200 lbm optimum design, and therefore its performance in the selected Basic Mission mode. This is shown in Table 3-1. However alternate missions can be postulated which have more effective performance because the axis-change maneuver of the Aeronomy/SS is not required. The saved propellant is partially applied to the 550T RCS requirements and partially to increasing ΔV capability. Sufficient time was not available to carry comparisons further, or to look at inertia wheels for fine RCS.

In summary, the Aeronomy/550T designs are competitive and may, on further analysis, reveal advantages.

5.1.2 Structure

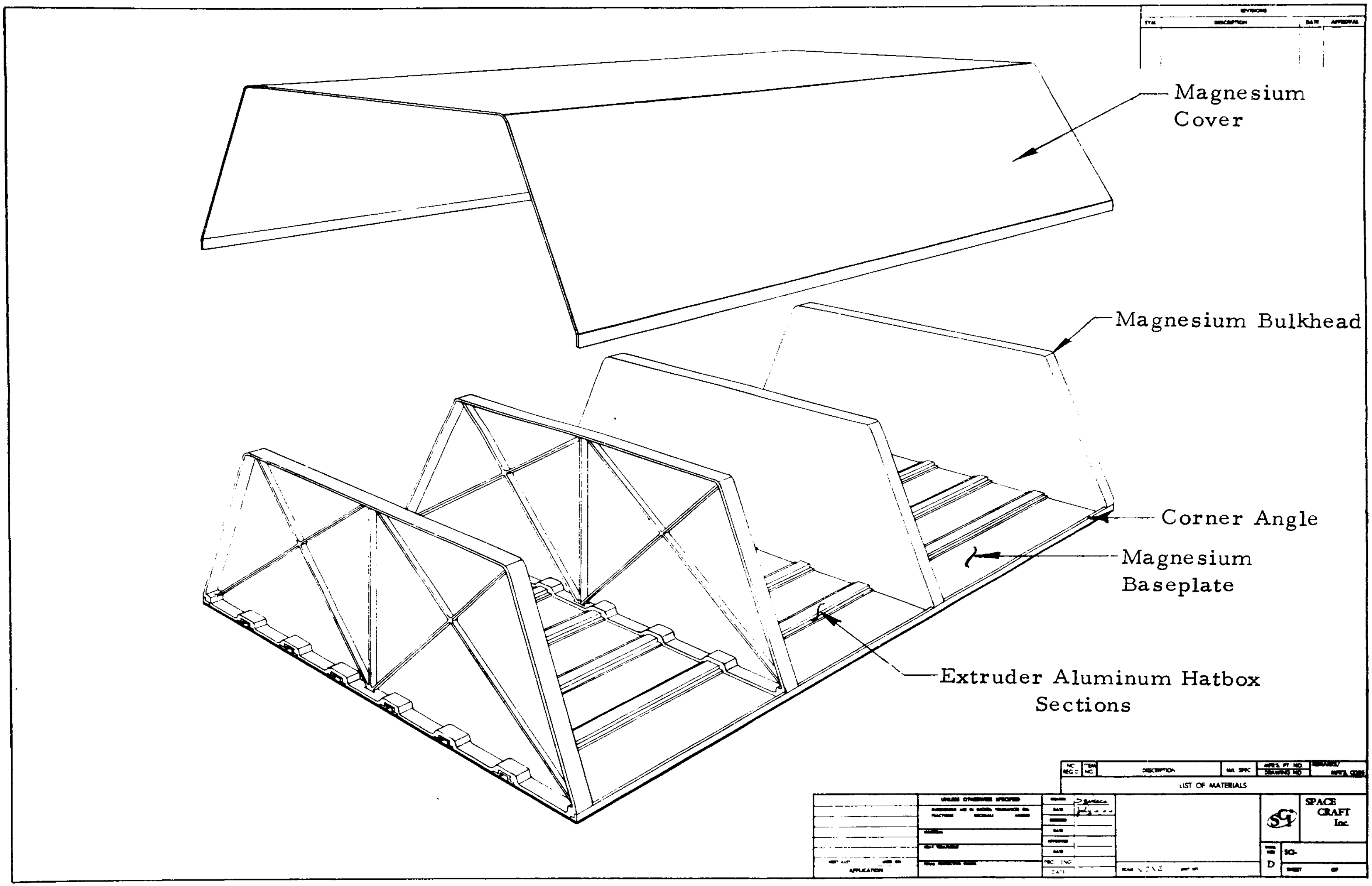
The need to reduce satellite mass to under 400 lbm for Scout launch required a concerted weight reduction attempt for all systems. The structural approach selected yields a mass of 47 lbm, or just over 12 percent of the 380 lbm allocated to the spacecraft less adapter.

Corrugated structures of fiberglass and aluminum were considered and rejected. A prime object of the design is to provide a rigid and accessible mounting surface for most systems. The structural concept for the Aeronomy/550 is shown in figure 5-4. A magnesium sheet forms the "top" of each of the two main body sections. This plate is reinforced with aluminum hat box channel against longitudinal buckling. Cast magnesium bulkheads are fitted to the plate over these channels to provide spanwise strength, mounting surfaces, and supports for the magnesium cover panel. The skin is fitted last and provides torsional stiffness. Only conventional fasteners were considered for the preliminary design. Quick fasteners can be considered in detailing the design.

The bulkheads have one smooth surface to facilitate mounting, and one ribbed face for strength. All are identical in the forward body section except for mounting holes etc.

The aft section contains the propulsion and RCS system, and may omit skin cover from the nozzle section of the unit to improve radiation cooling of the latter. Since the propellant tanks are short and stubby in the bipropellant Scout version, these are utilized to carry shear in the cross section plane, by providing proper cradles. The thrust distributing structure is essential to take up the nominal 100 lbf of the engine with minimum body deflections. The RCS module is based on the bulkhead structure and serves as rear support.

Both front and rear bulkheads of the satellite are adapted to mate with the Scout payload adapter, which will be modified to carry the wide, essentially split load. Modification consists of providing a suitable mounting surface, which should lie aft of the instrument projections in the case of the forward bulkhead.



AERONOMY/550S STRUCTURAL CONCEPT
TYPICAL COMPARTMENT
FIGURE 5-4

Mass balance considerations cause two sections of the battery pack to be moved to the aft compartment. Otherwise all communications, data and control electronics, as well as the experiments, are in the forward compartment to minimize cable runs. Flat cables are preferred to connect the two compartments.

Separation and deployment is activated when electrically fired squibs release restraining mechanisms. The major deployment of the folded-over halves of the structure into an extended flight configuration is shown in figure 5-1. This is activated after yo-yo despin and separation from Scout. A geared inertia train restraining the motion of a compressed spring will provide both governed action and positive latching. A torque motor can also be considered.

The solar panels are of approximately 1/4" honeycomb aluminum construction, and are activated periodically by an electric motor through a worm gear. The choice of materials and lubricants for the panel hinges is not made at this time, but recent information indicates that suitable choices are no longer a real problem.

The four 28" x 40" solar panels are provided with light edge stiffeners, and with vibration-damping separator buttons to prevent damage when folded. Panel wiring is accomplished by means of flat cabling. The flat panels readily accept large solar cell submodules.

Structural assembly begins with the 'top' plates and progresses to bulkhead integration - all systems are then installed before the panels and skin are attached. The 28" wide x 40" long compartments are easy to handle during assembly and test, with special fixtures needed only as an added convenience.

Typical weights and materials used are:

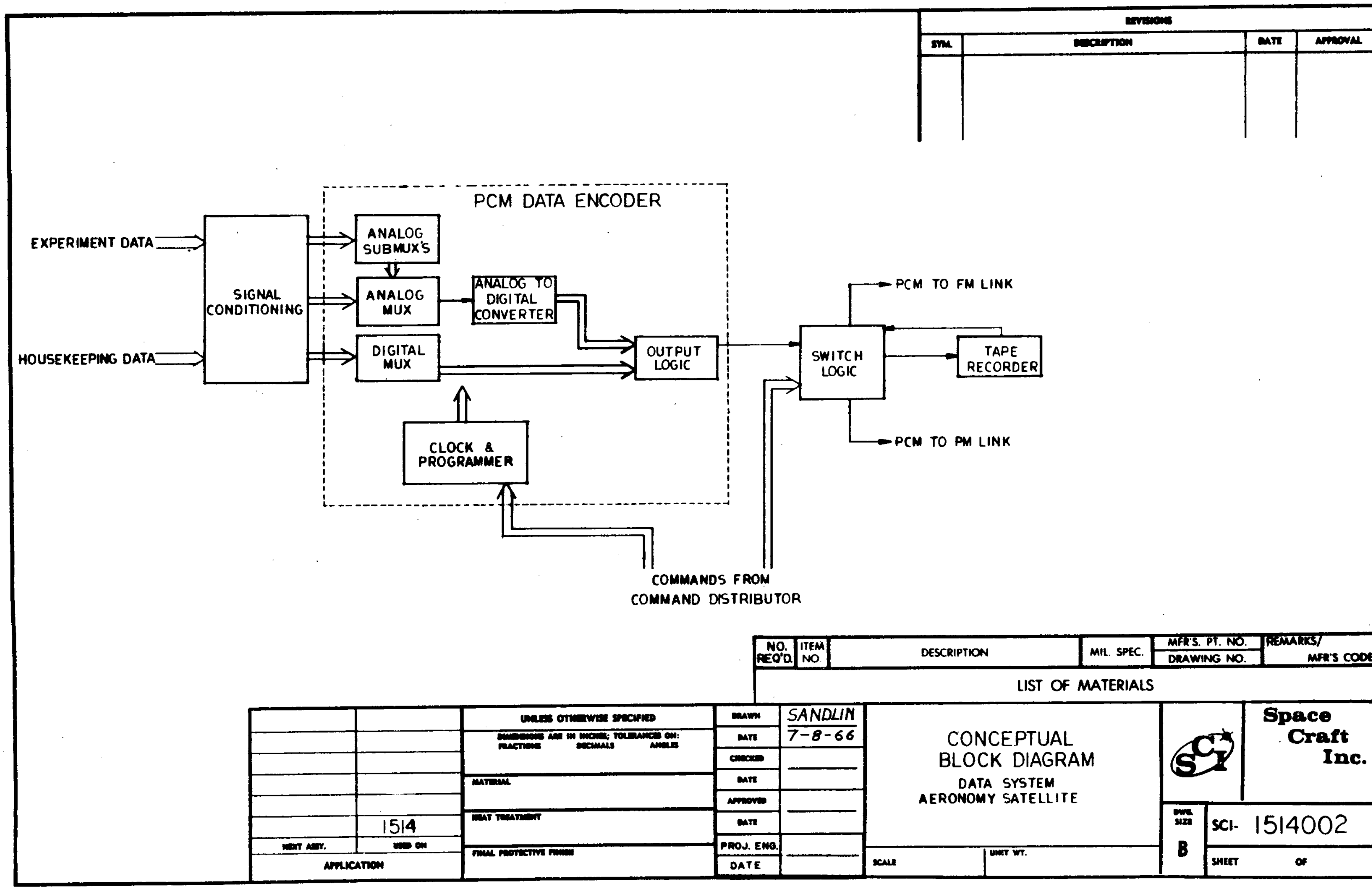
Magnesium base panel, 30" x 40" x .04" x .06 lbm/in^3 =	2.88 lbm
Magnesium Cover Panel, 46" x 40" x .03" x .06 lbm/in^3 =	3.31 lbm
Al Hat box Sections (9, req'd), 9 (40" x 1/12 x .103 lbm/ft) =	3.78 lbm
Al Angles 2 (req'd) 2 (40" x 1/12 x .4 lbm/ft) =	.63 lbm
Mg Bulkheads (4 req'd) castings at 2 lbs each =	8. lbm
Miscellaneous brackets, fasteners etc.	.4 lbm
Mass of one unit	19.0 lbm
Mass of both units	38. lbm

5.1.3 Data System

A recommended data system design for the aeronomy experiments was derived from a number of restrictions and assumptions based on available information from previous aeronomy missions and predicted regions of improvement desirability. The primary criterion was identified as an assumed desire to provide 100 percent orbital measurement coverage via tape recording and playback over selected ground stations. A cursory look at probable ground station coverage with STADAN indicated that at least one station per orbit could receive satellite data for a three minute period. A two and one half minute playback time was chosen for the recorder.

Recorder

An 88 minute record and 2.5 minute playback requirement gives a reproduce-to-record speed ratio of approximately 35:1. Endless loop recorders have nominal bit packing densities of 1500 per track inch (590 per cm.) and may record 4 to 8 tracks in parallel on quarter inch (0.635 cm.) tape. A



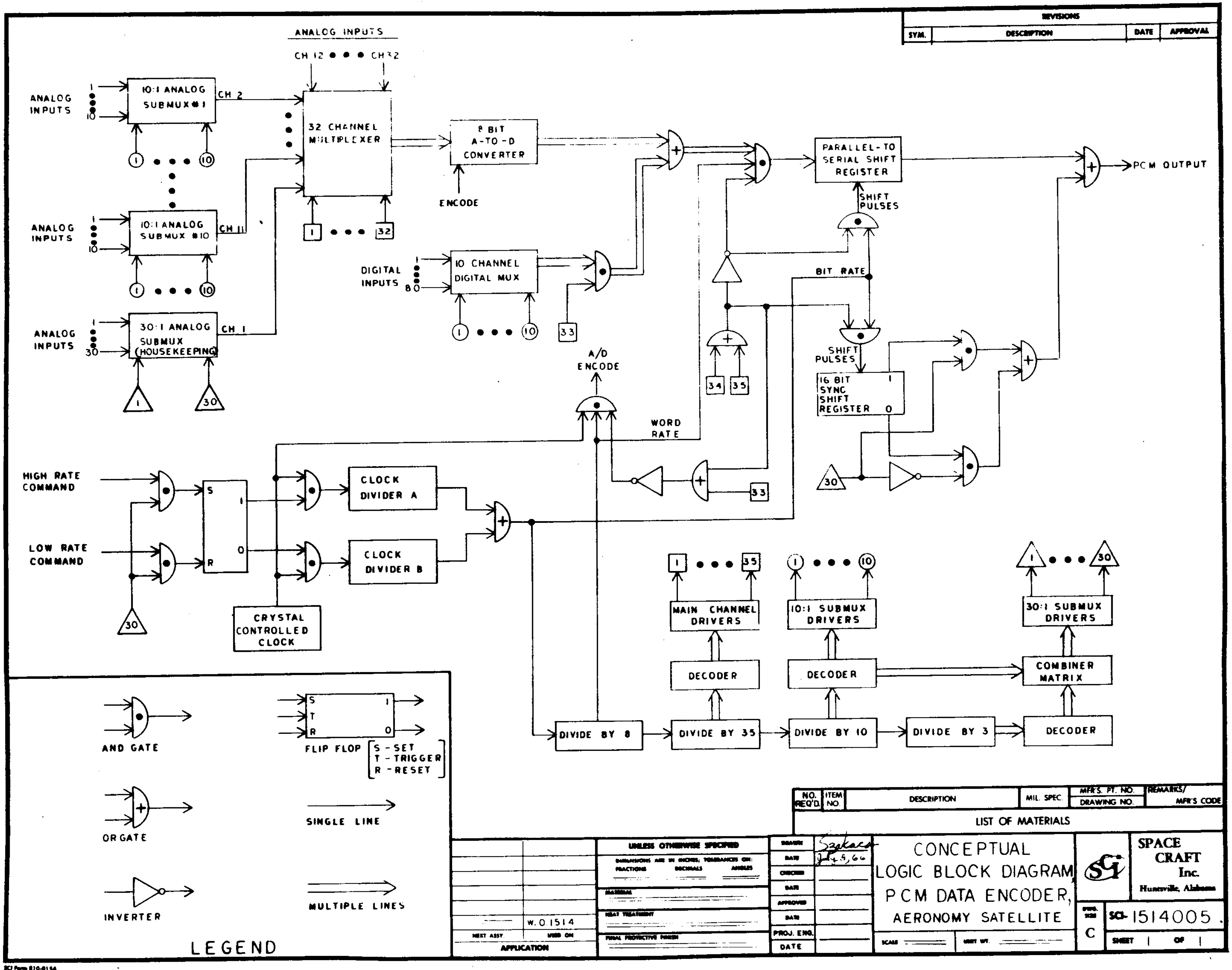
BLOCK DIAGRAM, DATA SYSTEM
FIGURE 5-5

tape reproduce speed limitation of 46 centimeters per second was recommended by one manufacturer. A recent NASA/MSFC contract award requires development of a 4 track, parallel, endless loop recorder operating at up to 590 bits per track centimeter (2360 bits per cm. total). This recorder appears compatible with the aeronomy requirements, and its restrictions, coupled with a reasonable bandwidth total of 300 kHz, radiated, gives a maximum bit rate of 100,000 bits per second from the recorder playback mode, or approximately 2800 bits per second into the recorder. With 8 bit resolution per measurement, 350 words per second may be programmed. Any number of data systems may be visualized which will operate in these requirements, and the one described below is a flexible system described merely to exemplify. A block diagram of the data system is shown in figure 5-5.

Encoder

Due to the non-spinning nature of the Aeronomy/550 satellite as opposed to previous aeronomy satellites, high sample rates to remove spin rate ambiguities are not required. The system described was derived from a basic 10 samples per second (sps) per measurement approach, giving a channel allocation of 35 slots. Cross strap options (supercommutation) may be exercised to increase sample rates to as high as 100 sps with a sacrifice to number of allowable measurements. Subcommutation options may be exercised for 10:1 subcom or 1 sps per measurement.

The approach, shown in figure 5-6 uses 35 time slots, two of which are used for synchronization, and one for digital discrete signals. Housekeeping and calibration data are contained in 30 measurements subcommutated onto one channel, leaving 31 channels available for experiment data. In order to keep the amount of wiring to a reasonable quantity, subcommutation options are limited by recommendation to 10 channels. Thus, there are 31 channels



BLOCK DIAGRAM, PCM DATA ENCODER
FIGURE 5-6

which may accommodate up to 121 experiment measurements, with all 10 optional subcommutation channels used. Two examples of how the system may be used are shown below.

Example 1: No Subcommutation

4 measurements at 40 sps (16 channels)
4 measurements at 20 sps (8 channels)
1 measurement at 60 sps (6 channels)
1 measurement at 10 sps (1 channel)
Total of 10 measurements on 31 channels

Example 2: Maximum Subcommutation

100 measurements at 1 sps (10 channels)
2 measurements at 60 sps (12 channels)
4 measurements at 20 sps (8 channels)
1 measurement at 10 sps (1 channel)
Total of 107 measurements on 31 channels

Obviously, many other options are practical. External switching at timed or commanded intervals may be incorporated to allow taking one set of measurements during one time period (e.g., one orbit) and another set during the next period. At encoder meeting the above requirements may be packaged into approximately 100 cubic inches (1639 cubic centimeters) at a weight of 3 pounds (1.4 kg) and a continuous power consuption of 2 watts. Clock divider chain switching can provide the maximum 100,000 bits per second for real-time or 1:1 recorder operations, if desired. This clock switching would provide 35 times the normal sampling rate discussed above; i.e., a normally operated measurement of 19 sps would be sampled at 350 sps and a 60 sps measurement would be sampled at 2100 sps. With a small amount of programming, high rate data could be recorded for 2 1/2 minutes over a specified latitude bracket and played back over a ground station at a later time.

5.1.4 Communications System

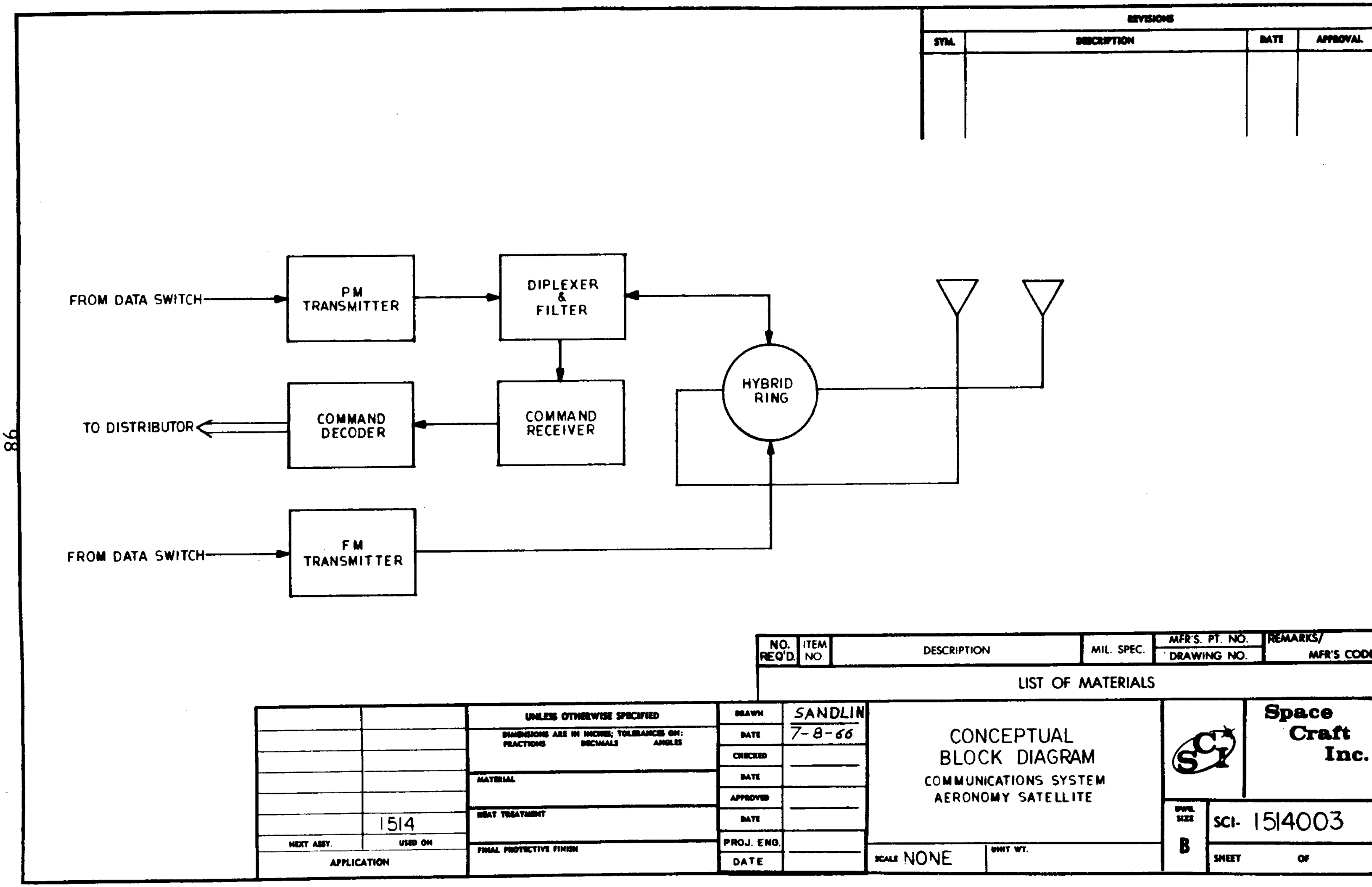
The communications system described in this report, and diagrammed in figure 5-7, operates to give continuous real time transmission of data by means of low power phase modulated transmitter, also used as a tracking transmitter with Minitrack ground equipment. A second transmitter utilizing frequency modulation is used to transmit the data which has been stored in the spacecraft's magnetic tape recorder. Due to the large ratio of tape-reproduce to tape-record, the FM transmitter is required to radiate at an increased bandwidth and consequent higher power. Standard satellite transmitters compatible with the STADAN frequencies of 136 MHz to 137 MHz are readily available with several choices of bandwidths. STADAN data acquisition receivers have selectable predetection bandwidths of 10 kHz, 30 kHz, 100 kHz, 300 kHz, and 1 MHz. Consideration of restraints imposed by data system limitations, power consumption, procurability, and interactions of frequencies led to a choice of 300 kHz as the design point for the high rate data. With proper separation of transmitter frequencies, a 30 kHz bandwidth may be used with the low power PM transmitter.

The satellite operates at approximately 100 N. miles (185 km) with a maximum slant range, at a five degree elevation above station horizon, of 550 N. miles (1020 km). The required transmitter power at this slant range is given by the following equation.

$$10 \log_{10} P_t = L_p + L_{fs} + S/N + N.F. - G_r - G_t + P_n + B$$

where	L_p	=	Polarization and insertion loss	8.5 db
	L_{fs}	=	Free space loss	
		=	$32.9 + 20 \log f + 20 \log R$	
			F = MHz = 137	
			R = Kilometers = 1020	135.8 db
	S/N	=	Required predetection signal to noise ratio	15.5 db
	N. F.	=	Receiver noise figure	3.5 db
	G_r	=	Receiving Antenna gain	19.0 db
	G_t	=	Transmitting Antenna Gain	0.0 db
	P_n	=	Average sky and noise temperature	
		=	10 log kT	
			$K = 1.38 \times 10^{-23}$ joules/^{o}Kelvin	
			$T = 290^{o}$ Kelvin	
		=	-204 dbw per cycle of bandwidth	
		=	-174 dbm	-174.0
	B	=	Predetection Bandwidth	
		=	10 log 300 kHz = 54.8 db	
		or	10 log 30 kHz = 44.8 db	

Thus for the high power transmitter, 10 log P_t = 25.1 dbm and P_t = 324 milliwatts which provides 4.9 db safety margin when a minimum transmitter power of 1 watt is used. The low power transmitter requires 10 db less bandwidth or 32.4 milliwatts, with the same safety factor when using 100 mw as the transmitting power.



BLOCK DIAGRAM, COMMUNICATIONS SYSTEM
FIGURE 5-7

Command System

The command system to be used with the aeronomy satellite is compatible with the STADAN system and requires a receiver and information decoder. The command format used by STADAN is described in GSFC report: Tone Digital Command Standard, dated January 15, 1963, and is summarized in figure 5-8. The command receiver is diplexed with the low power PM transmitter. Channel separation in the diplexer is 20 db minimum; additional filtering can be provided in the diplexer for sufficient attenuation to satellite transmitter power in order to prevent receiver desensitzing.

Antennas

A magnetic turnstile antenna (crossed loop) is used for radiation and reception. The antenna is recessed in the metallic skin to prevent the addition of cross sectional aerodynamic drag area. The antenna is driven by dual outputs from the hybrid ring. Phasing is accomplished with the cables feeding the antenna from the ring. Networks at the terminal of the antenna match the antenna impedance to the impedance characteristics of the feed cables. The system has essentially zero gain with nearly hemispheric pattern coverage. Right or left polarization may be selected. Figure 5-9 is a view of one loop antenna; the second loop would cross through the antenna at the tuning network.

5.1.5 Power System

The equipment orbital power average is 22 watts providing 44 watts to the experiment. A 20 AH capacity battery (50 lbm) was chosen for this application. The nominal battery discharge period is assumed to be 53 minutes, leaving 35 minutes of solar operation to recover the battery and power the load. A recovery requirement of 1.4 times the energy taken out of the battery is used to determine the replacement energy. Further detail of the power system is given in Appendix C.

The command format[1] utilizes a pulse duration coded format which 100% amplitude modulates an assigned tone frequency. The modulated tone 75% amplitude modulates the assigned carrier. The frame format of the coded signal contains six blank bits, six sync bits, two address codes, and three command codes as illustrated in Figure 3-23.

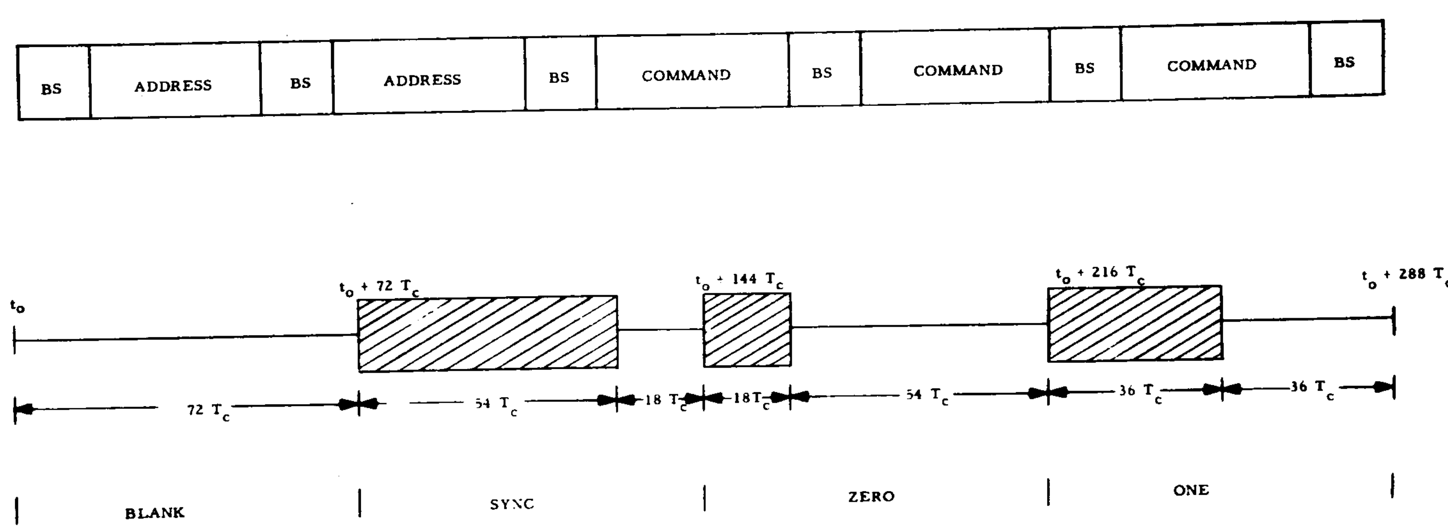
The address word is used to "unlock" the decoder under command; it is a unique 10 bit word to which only one decoder will respond. The bit composition is:

Bit 1 = blank
Bit 2 = sync
Bit 3-10 = 1 and 0 coded bits.

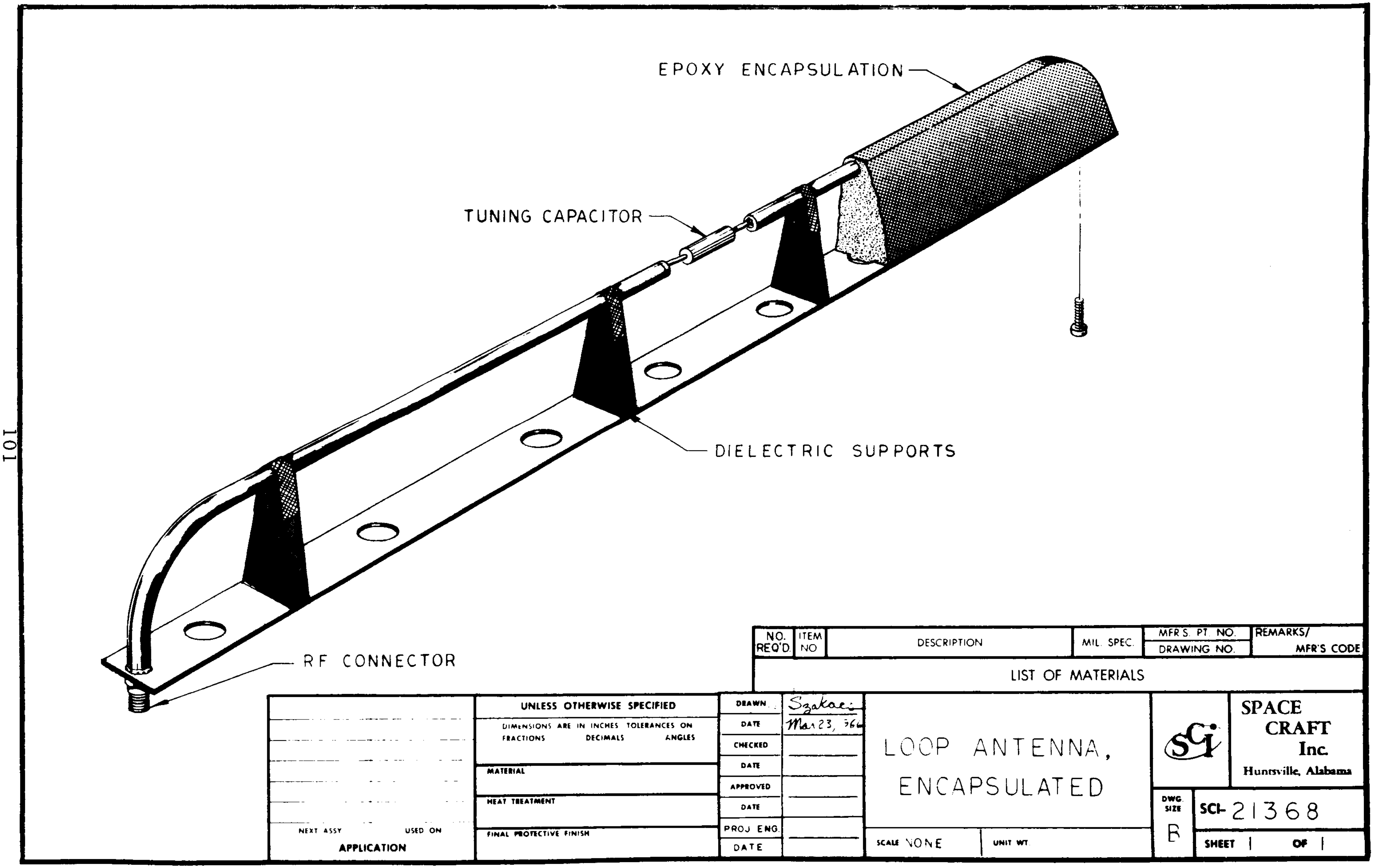
The command word is a combination of bits to which the decoder, once addressed, will respond and output voltages on discrete lines. The bit composition of the command word is:

Bit 1 = blank
Bit 2 = sync
Bit 3-6 = first binary word
Bit 7-10 = secondary binary word

[1] GSFC Report; Tone Digital Command Standard; January 15, 1963.



COMMAND FORMAT
FIGURE 5-8



LOOP ANTENNA
FIGURE 5-9

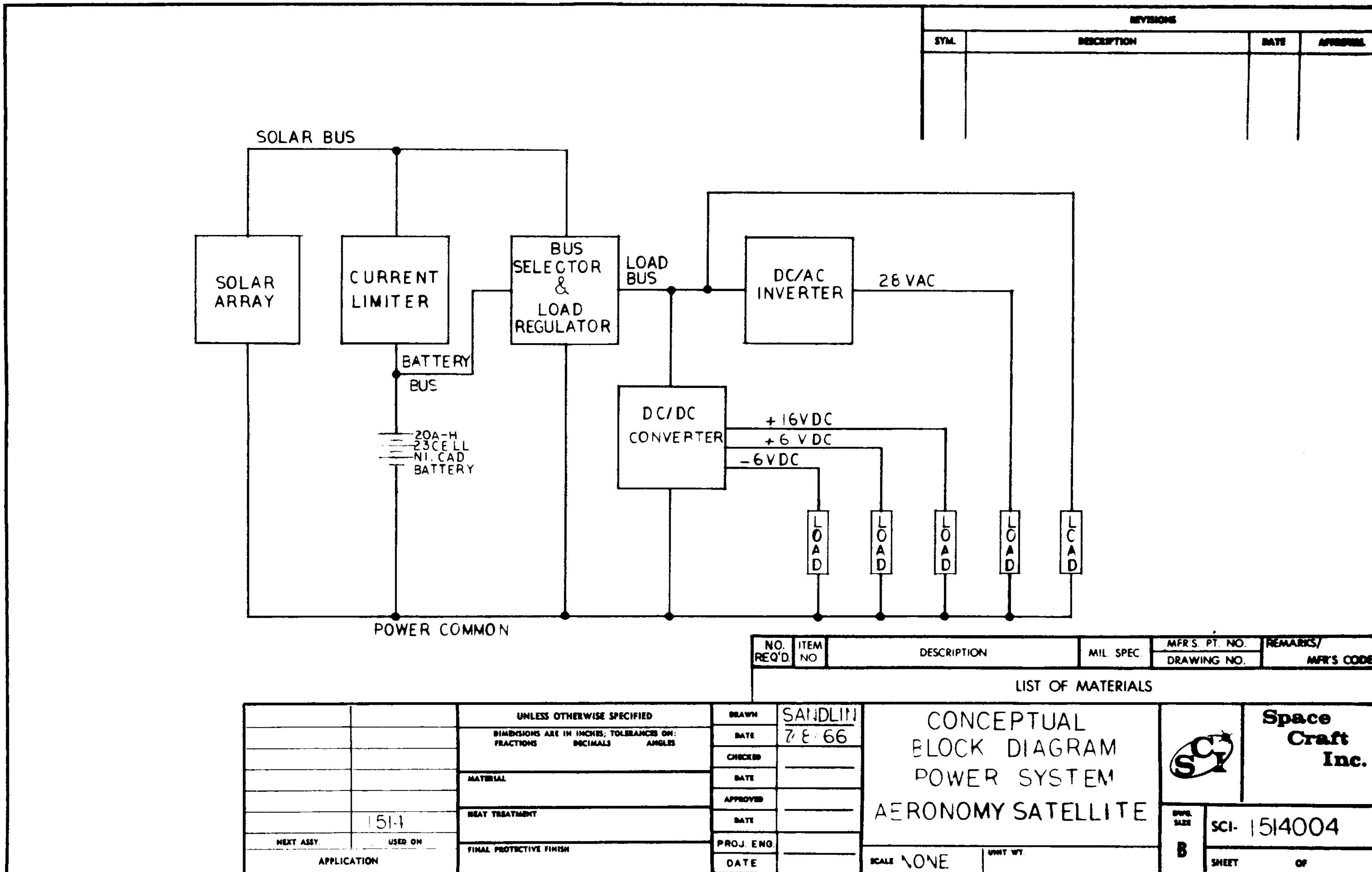
Battery

Losses normally associated with battery charging (charger inefficiencies) are minimized by allowing the battery voltage to set the solar bus voltage and eliminating the series charger. This approach allows all excess current to be dumped into the batteries and requires that the actual capacity of the battery be maintained substantially below its maximum rating, since current must be strictly limited into a fully charged battery. If the battery is properly maintained with respect to its capacity rating, a c/2 charge rate may be applied directly, provided that a maximum capacity of 75 percent of rating can be assured when operating at maximum sunlight durations. Protective parallel regulation should be provided to assure that cumulative charge build-up does not raise the battery capacity to such an extent that the high charging currents would damage the battery. The charging system described above is not recommended when long battery cycle lifetimes are required, but it is completely satisfactory for the short lifetimes referenced here; in fact, the same concept was used by early Juno II Explorer satellites and provided successful battery operation for over six months (Ref: NASA TN D-608; Juno II Summary Project Report, Volume I, Explorer VII Satellite). A block diagram of the power system is shown in figure 5-10. The mass breakdown of the solar panels if given below:

a) 1/4" honeycomb sections of .013" aluminum skin and .002" aluminum honeycomb; 0.486 lbm/ft^2

b) glassed cells; 0.213 lbm/ft^2

c) total panel mass (4 req'd); 4 (40" x 30" x 1/144 x 0.7) = 23.30 lbm



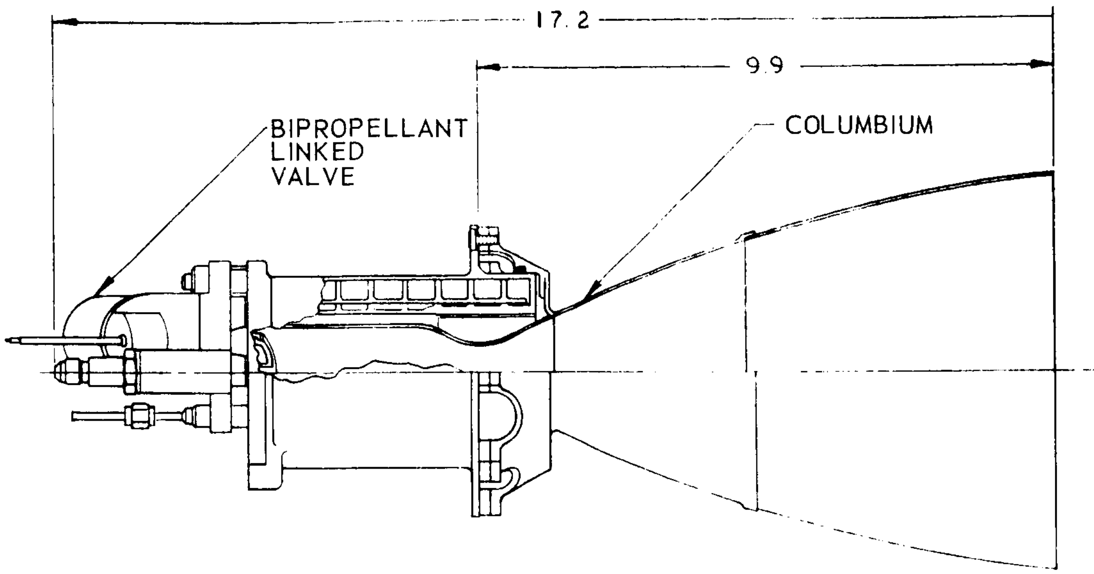
BLOCK DIAGRAM, POWER SYSTEM
FIGURE 5-10

5.1.6 Propulsion

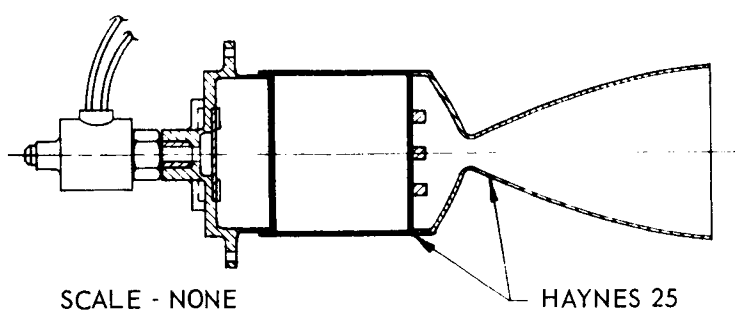
The propulsion system for the Aeronomy Satellite consists of a C-1 engine, capable of multiple restarts, fed by two N_2 pressurized tanks through dual redundant and quick acting valves. Figure 5-11 shows the typical engine sections useful for this application. The pressurization system also has redundant valving and means for unloading the regulator during extended engine shutdown. The propellant to be used is N_2O_4/UDMH which does not require hypergolic slugs or catalyst decomposition beds. The propulsion system diagrams for a monopropellant and bipropellant system are shown in figures 5-25 and 5-26. The system provides minimum impulses of 0.89 Newton-seconds corresponding to a velocity correction capability of 0.004 meters per second. Three modes of operation are available in the propulsion: ground controlled; automatic single burn; and automatic dual burn. The paragraphs which follow are summaries of material presented in the final report prepared by SCI for MSFC under NAS8-20253. In the ground controlled mode, the engine is started and stopped via the ground command link. The delta-velocity is controlled by the engine thrust and the duration between start and stop commands. In the single burn mode, the desired velocity increment and the time of initiation is transmitted to the spacecraft. A linear integrating accelerometer is used to compare actual velocity to commanded velocity and to shut down the engine when the proper rate is reached.

The dual burn mode operates the same as the single burn, except that two velocity increments and two initiation times are commanded. The purpose of the dual burn is to allow automatic Hohman orbit transfers.

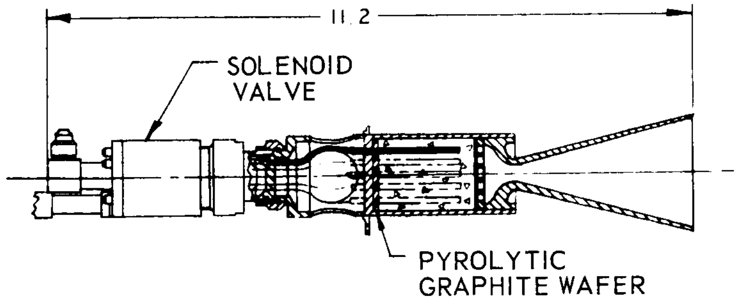
Thrust vector alignment is provided by a jet vane control loop depicted in figure 5-12 . The control loop operates in conjunction with the autopilot A.C.S. functions described in the following subsection.



A. Nominal 100 lb f Bipropellant Engine



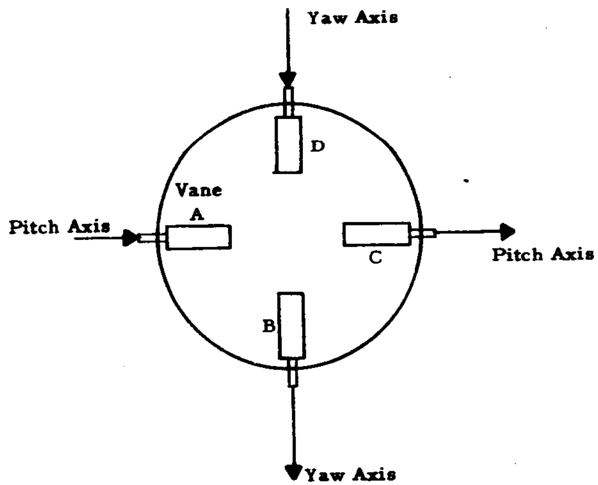
B. Nominal 250 lb f Monopropellant Engine



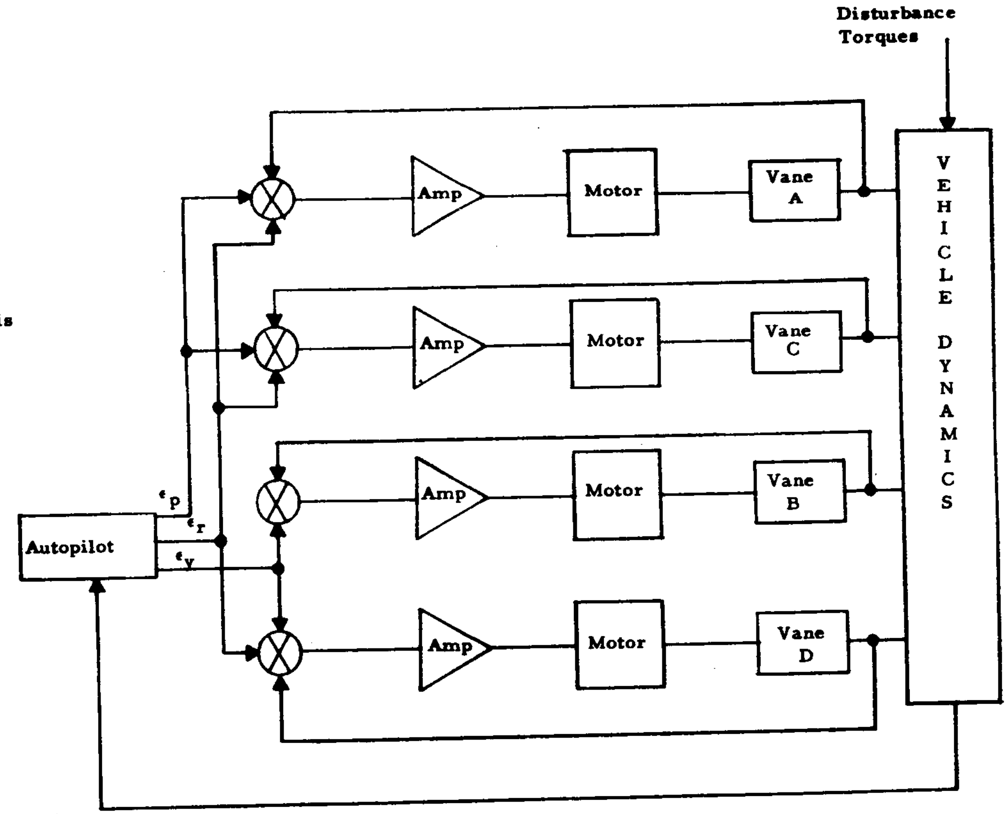
C. Nominal 20 lb f Monopropellant Engine

TYPICAL ENGINE SECTIONS

FIGURE 5-11



Pitch Control	Vanes A and C
Yaw Control	Vanes B and D
Roll Control	Vanes A, B, C, and D



BLOCK DIAGRAM, JET VANE CONTROL LOOP
FIGURE 5-12

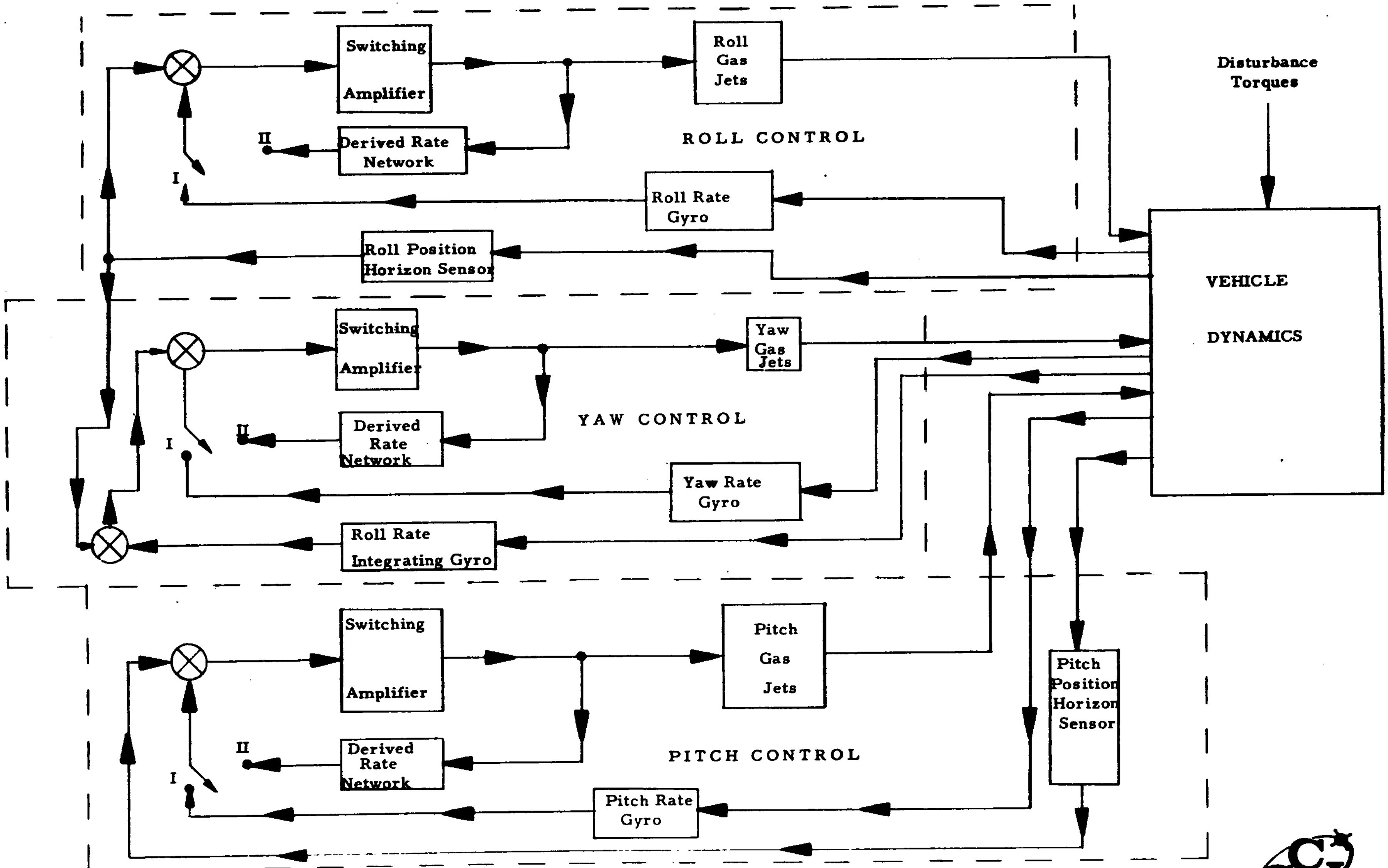
5.1.7 Attitude Control

The attitude control requirements are met with two major functional systems: cruise and autopilot, both of which operate through the A.C.S. computer. The cruise function, or mode, utilizes a horizon sensor for roll, a horizon sensor for pitch, and gyrocompass to provide yaw information. The sensors provide angular information to the computer, which commands the reaction control jets in a limit cycle based on derived rate information. A block diagram of the attitude system is shown in figure 5-13.

The autopilot functional system operates in two modes: acquisition, and propulsion. These modes require three rate gyros. An integrating accelerometer is used to provide delta-velocity information during thrusting.

The acquisition mode is used for initial reference, and for reacquisition at the end of propulsion maneuvers. The rate gyros are used to null the vehicle rates and in conjunction with the horizon sensors are used in a search rate to establish pitch and roll about the local vertical. The yaw gyro is switched into the control loop to establish the flight vector.

Commanded turns which are optional are accomplished by two rotations; yaw direction turns and established by commanded yaw angles executed at a constant rate and compared with yaw positional information until proper match is obtained. Once the yaw position is established, it is maintained and the horizon sensors are switched out and gyro control is initiated on the pitch and roll axis, and a pitch maneuver is initiated and continued until the vehicles attitude is in conformance with the commanded values. When the turn is accomplished, the position is maintained in a strap-down guidance mode.



BLOCK DIAGRAM, ATTITUDE CONTROL SYSTEM
FIGURE 5-13

During propulsion, the autopilot provides rate and position error information to maintain the established orientation. Thrust vector alignment is maintained by the autopilot through the propulsion jet vane control loop illustrated in figure 5-12.

The reaction control system is a mass expulsion type using gaseous nitrogen as the working fluid. The system uses eight thrusters: two for yaw control, two for pitch control, and four for roll control. The thrusters are mounted on a plate at the aft end of the satellite.

5.2 AERONOMY/SS APPROACH

The Aeronomy/550T design was 90 percent complete when more explicit Goddard Space Flight Center requirements were obtained. The remainder of this study essentially develops system details used to back up the parametric analyses of the spin stabilized Aeronomy/SS concept. This design eliminates articulated solar panels, rotating-prism horizon sensors, and gas-jet limit cycle reaction control system. It incorporates a two-pulse scheme for precessing the thrust-spin axis, and spin momentum dump jets in a flat cylindrical shape. The shape requires that during low altitude, high drag conditions the cylinder be precisely controlled to orbit "edge-on".

The data handling system is based on GSFC experience and provides a desired combination of data resolution and instrument angular position resolution, for intermittent periods governed by available power. Within predictable limits there are no 'critical' systems in this design, although both ACS and propulsion require detailed analyses. Configuration restrictions are loose, but weight reduction in structure and propellant tanks improves system performance.

5.2.1 Configuration Analysis

The spin stabilized aeronomy satellite is found to have a conventional configuration for such systems, except that it is more disc-shaped than usual. This is due to a desire to minimize drag, and although this is not a strong criterion it can be met without compromising other requirements. The principal one of these is to package a maximum of propellant with least inert mass expenditure. Sections 5.2.2 and 5.2.6 show that neither satellite nor propellant tank geometry influences mass enough to dictate shape for preliminary design purposes. This leaves the criteria to be considered those of experiment packaging flexibility and minimum drag.

Design Limits

Before discussing specific configurations, it is well to assess the effect of various constraints on system parameters.

The system requirements set a lower level of 5 watts to the experiment. In Section 5.2.5, this is shown to require a solar cell projected area of 3.75 ft^2 for a cylindrical shape. Therefore a practical lower limit to size is set. An upper diameter limit is set by the Thor shroud constraint, given as 5 feet. Furthermore, launch weight limits of 740 lbm and 1250 lbm are given by a minimum-stage Delta and an improved long-tank version with strap-on solids, respectively. At the outset, spacecraft adapter weights were estimated and net payload weight targets of 700 lbm and 1200 lbm were set for the two design limits.

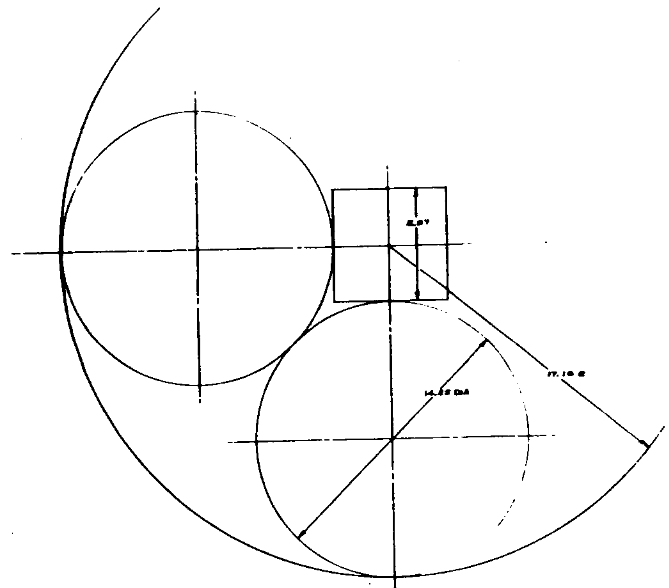
Principal Configuration Parametrics

The freedom to choose configurations within fairly wide limits is established in the next section. This fact permits concentration on the all important propellant mass fraction of the satellite, which determines directly the velocity increment that can be realized by the system. In other words longest lifetime is achieved by storing maximum energy on board, within given mass limits.

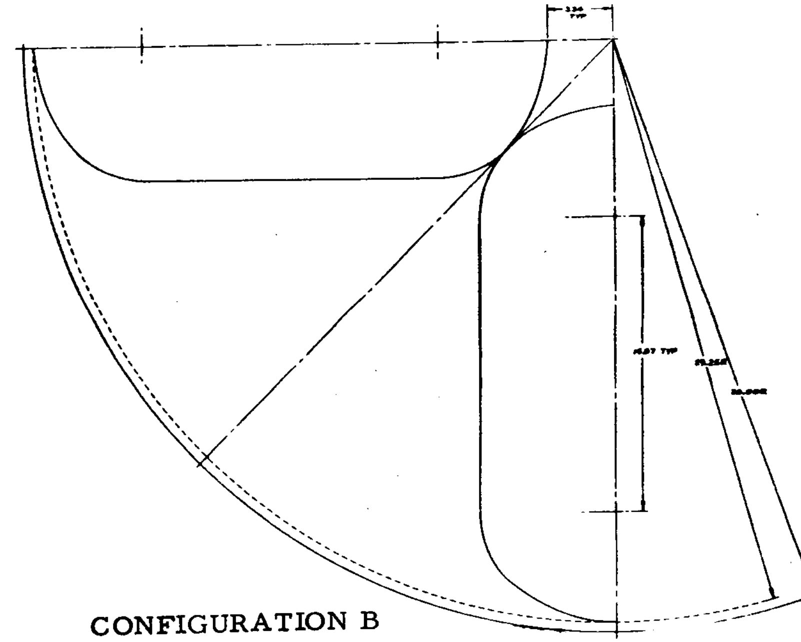
The assumptions made regarding structural factors for the satellite structure and the propellant tanks are collected in Section 5.2.6, and the resulting relations between the propellant mass and the total, initial satellite mass have been presented as systems criteris independent of configuration in Section 3, figures 3-1 through 3-4. To limit the possible variations, only the combinations: optimum tanks with optimum structure, and target tanks with target structure, have been plotted. For other combinations, the lifetime tables of Section 4 may be entered with propellant mass values between those of the optimum and target values of figures 3-1 for a hydrazine system and 3-2, for a bipropellant system.

Tank Layout

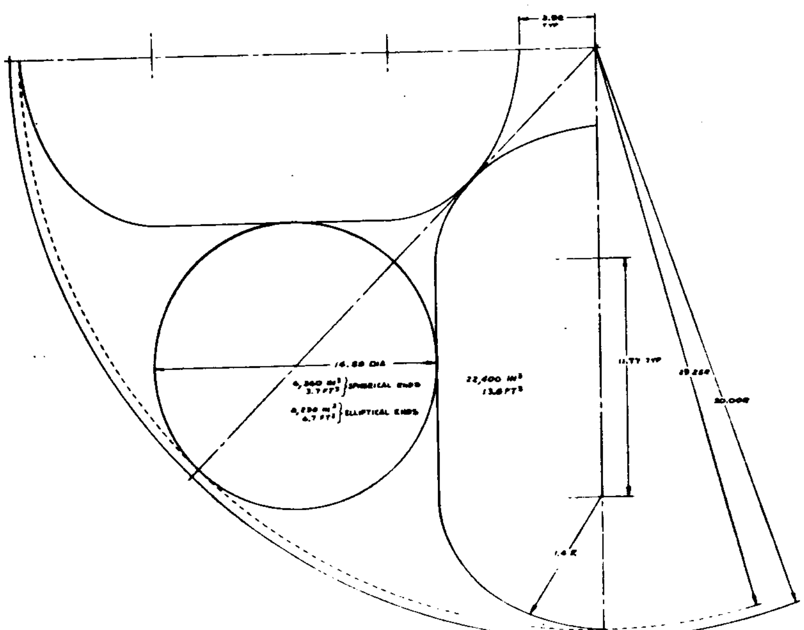
A series of point-designs were laid out for the 1200 lbm satellite. Evolution of approach is shown in figures 5-14. The effort was aimed at packaging the most propellant mass behind the least frontal area, and/or in the lowest total volume. It can be rapidly established that optima exist for various packaging criteria - it is not immediately clear which criterion is best for the satellite mission, or indeed which of a series of configurations most nearly approaches a given optimum. Tables for use in preliminary design are given in Appendix B, together with a discussion of various shape factors and their effects. It is sufficient here to state that the ratios tank volume/satellite volume, and tank volume/projected area increase with progression from spheres (refer to figure 5-14) to elongated tanks with hemispherical ends; to various nested tank configurations; and finally to the close-nested ellipsoidal head tanks shown in figure 5-15.



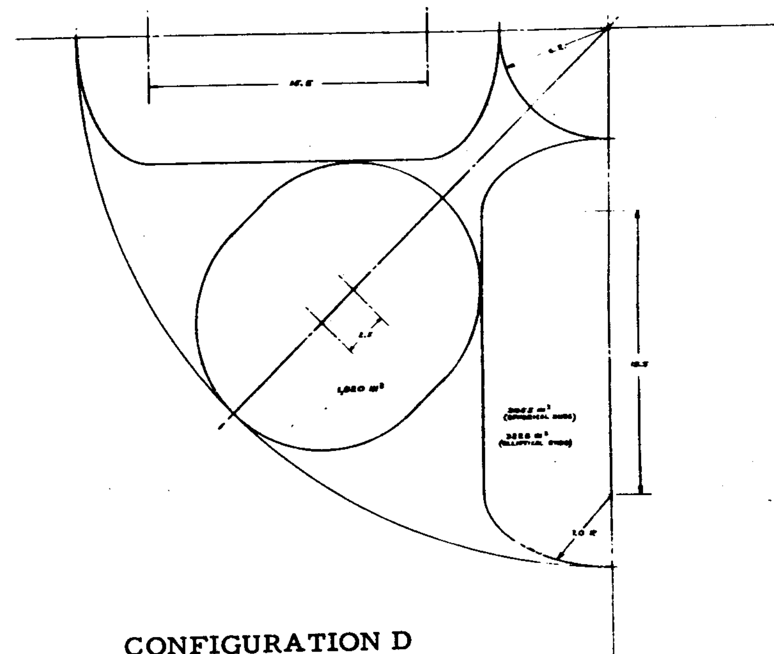
CONFIGURATION A



CONFIGURATION B

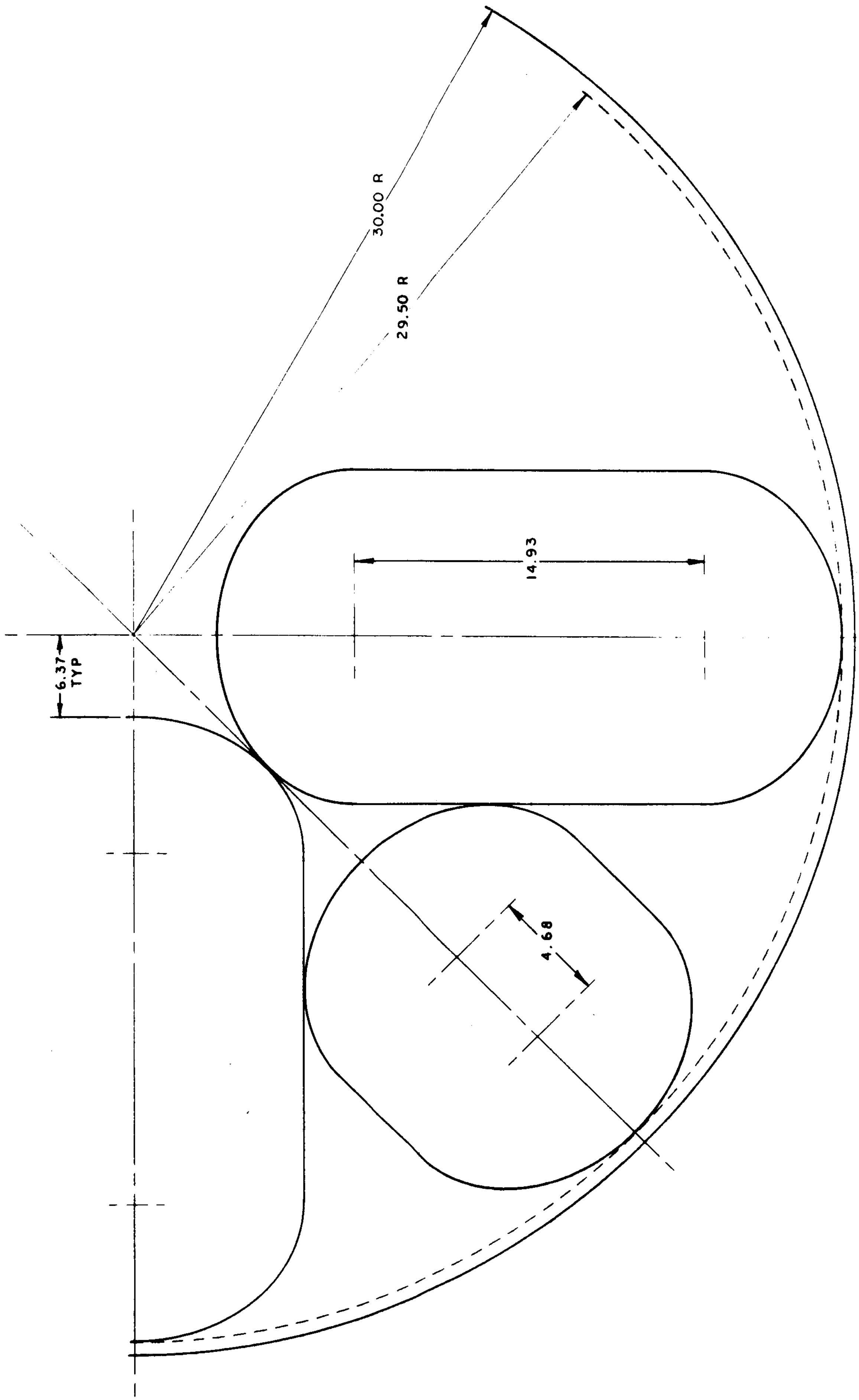


CONFIGURATION C



CONFIGURATION D

TANKAGE CONFIGURATION
FIGURE 5-14



TANKAGE WITH ELIPTICAL ENDS
FIGURE 5-15

Table 5-2 characteristics summarizes applicable design taken from Appendix B. WP is the hydrazine mass, W the total tank system mass for N plus N nested tanks. The table ranges over about 10 percent of the allowable masses. Figure 5-16 summarizes the range of designs investigated for the 'optimum' satellite. Four lower curves give WP, four corresponding upper curves give W. The optimum design mass limit is shown, as is the effect of the 2.5 ft radius in limiting design choice. The width of each N-band represents a choice of dimensional ratios.

Subsystem and Experiment Packaging

It is established therefore that configurations similar to figure 5-15 will satisfy the basic satellite requirements. Many variations are of course possible within the constraints of N=3, 4, 5. In practice these will probably be addressed to creating desirable experiment packaging configurations - the satellite subsystems can be packaged in the interstices between tanks, but given experiments may require more room. The actual space for experiments is quite large, however, and with moderate ingenuity the arrangement is usable as shown. Reduction to N=3 (six tanks) allows more interstitial room at slight drag expense for equal mass ratios. Note that sensors such as the recent quadrupole mass spectrometer designs can be accommodated on the satellite's equator.

5.2.2 Structure

The object of this section is to provide mass estimates representative of a realistic structural design approach.

Systems analysis indicates that the goal of one-year mission life can be accomplished only with relatively tight design. The underlying argument is that lifetime is proportional both to the ballistic coefficient W/C_dA and to the

OPTIMUM DESIGN (TABLE BI)

Wmax = 1200 - 329 = 871

WP	N	A	G	W
667	4	6.	1.5	862
647	5	5.5	2	837
630	3	6.	2	815
621	4	5.5	2	802
616	3	6	1.5	795

TARGET DESIGN (TABLE B2)

Wmax = 1200 - 403 = 797

566	4	5.5	1.5	796
548	5	5.	2.	770
525	4	5.	2.5	738
514	3	5.5	2.	723
514	4	5.	2.	722

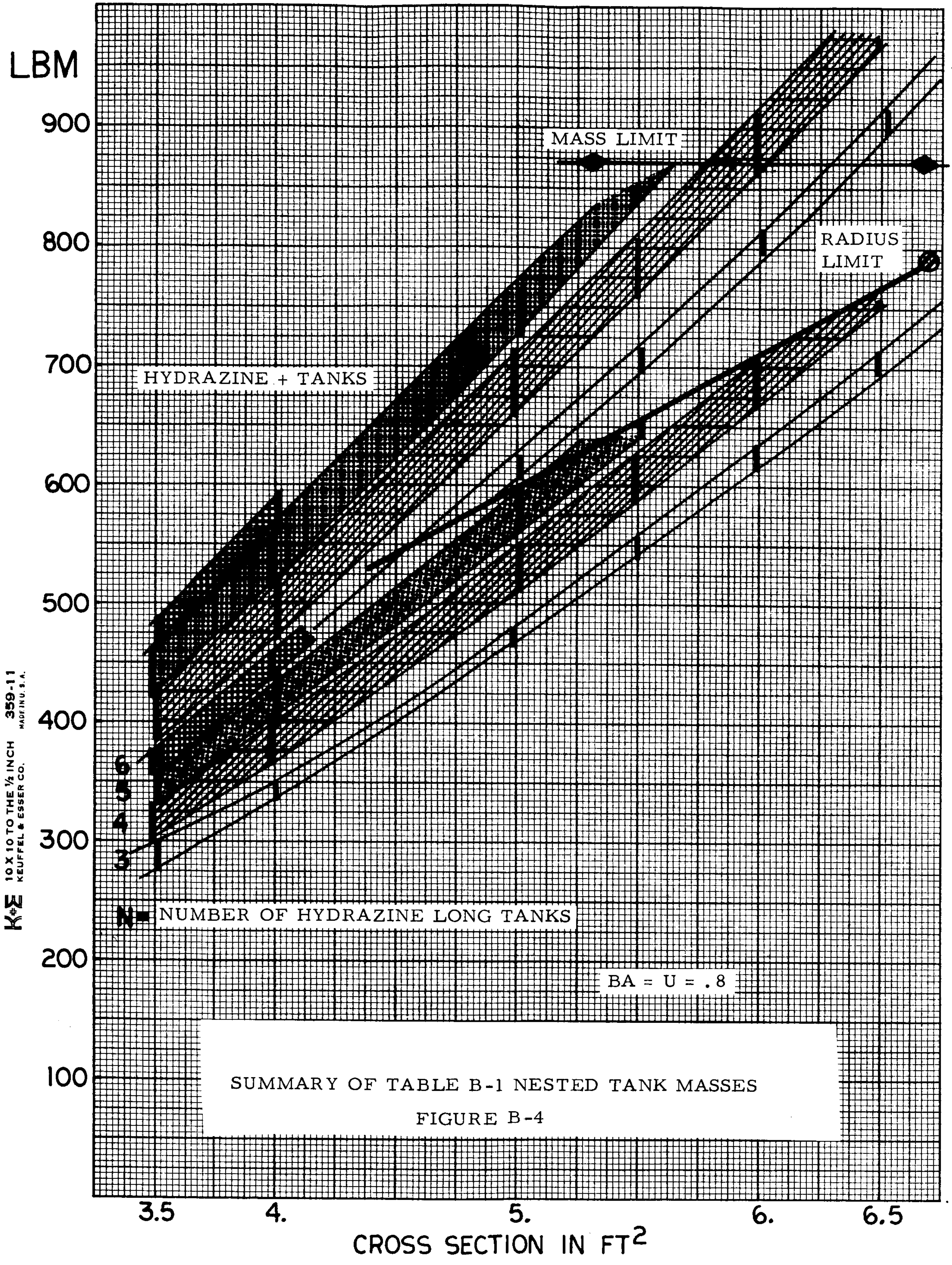
700 DESIGN (TABLE B2)

Wmax = 700 - 289 = 411

287	4	3.5	1.5	404
272	5	4	1.5	387
262	3	3.5	2.5	368
260	3	3.5	3.	366
256	3	3.5	1.5	360

SHORT TABLE OF TANK CONFIGURATIONS

TABLE 5-2



SUMMARY OF TABLE B-1 NESTED TANK MASSES

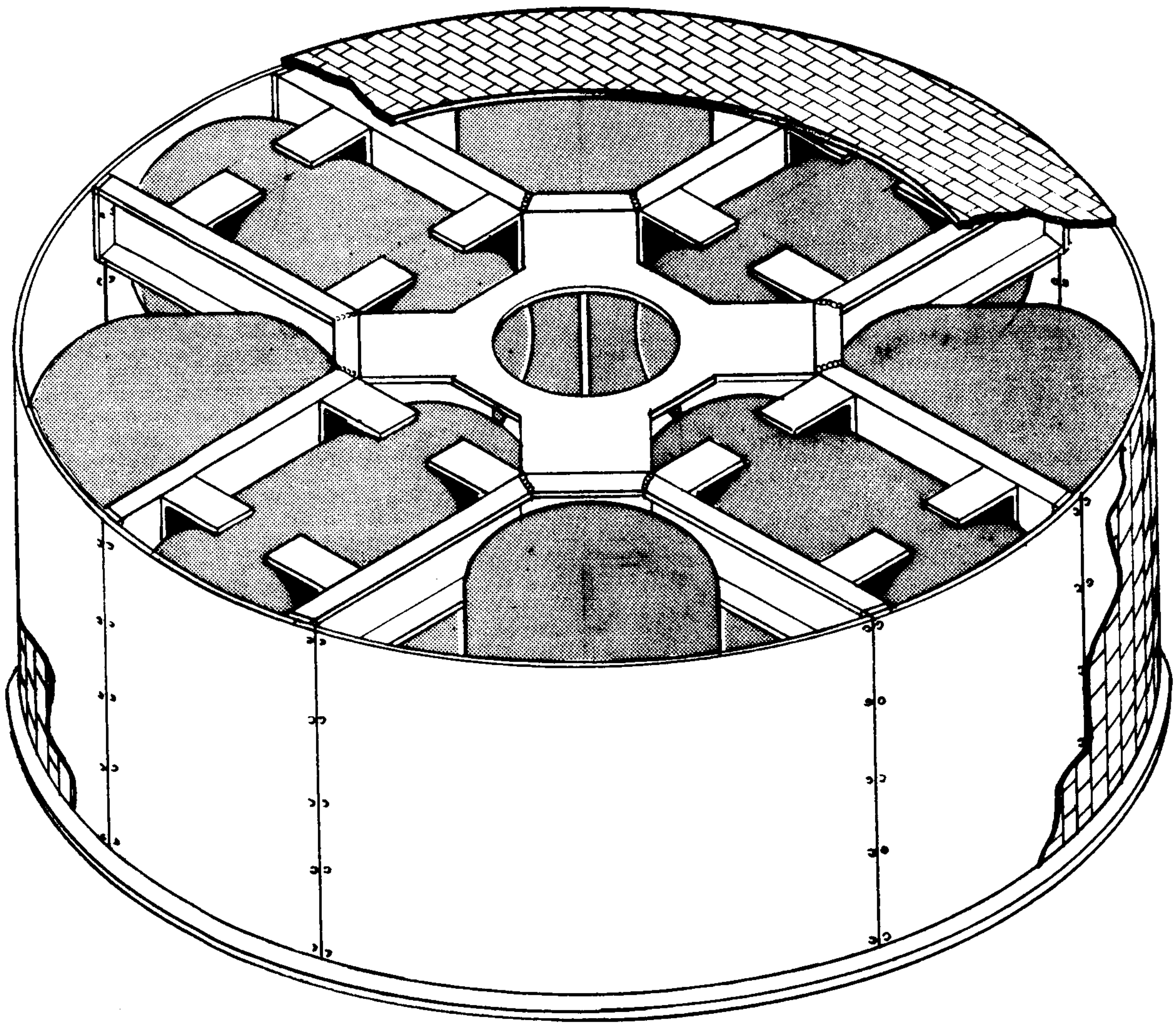
FIGURE B-4

amount of propellant available. Here W is the total mass at any instant, and A is the satellite frontal area. Propellant tankage configuration analysis leads to eight tanks, four longer and radially places around the engine, and four shorter rested in the interstices. To maximize W/A, the tanks should effectively butt. This configuration leaves little room for the required structural members.

The total lift-off mass of the parametric variations on design will remain constant (either 700 lbm or 1200 lbm depending on lift vehicle option). A realistic design approach of one size will therefore serve to estimate structural mass, even though slight changes in satellite mass are parametrically introduced. The maximum allowable satellite radius is demonstrably the best choice to minimize W/A with the selected tank configuration; and the major masses of the design are radius-dependent.

The indicated design of figure 5-17 uses a central hollow hub to take up G-loading and engine thrust; and supports each tank in four parallel beams fastened to end plates of the column. The tanks serve to prevent shear since they lie in cradles fastened to the beams. The outboard beam ends are joined to two rings which form the upper and lower outside edge of the satellite. Solar cells are affixed to top and bottom hollow circle sections which also act as rim stiffeners; and to the radial faces. These faces are separately removable from between the stringers connecting the upper and lower rims, and provide equipment access.

Tanks will be mounted after the skeletal structure is assembled, and will be integrated into the structure by clamping their cradles to predetermined torque. The solar panel sector fitted over the tank end may provide additional restraint if needed.



AERONOMY/SS, STRUCTURAL CONCEPT

FIGURE 5-17

The problem of transmitting engine thrust to the satellite rim without benefit of solid transverse beams can thus be overcome for the least materials mass by using the tanks as described. Launch thrust is distributed to both the center hub and the rim by a modified launch vehicle adapter. Fifty lbm is reserved for this item.

Other designs considered were based on resting the long tanks in hollow box beams, and carrying the through-beam load at their intersections; and variations of the suggested scheme which carried some beams across the edge of the hub to improve diametric integrity. Honeycomb was also briefly investigated. Either structural factors or access appeared less attractive, therefore the design described above is used to derive a structural mass estimate.

Structural Mass Estimates

A structural weight of 150 lbm was derived for a 5 ft. diameter/1200 lbm satellite case, using conventional aluminum materials. This gives a structural factor

$$k_{str} = \frac{150}{1200} = .125$$

Shape factors in the region of interest for this 1200 lbm design point are essentially constant for the structure described: that is, the sum of satellite radius and tank radius, which governs much of the estimated stringer and skin masses are nearly constant. The structural factor of .125 is therefore accepted for the target design in the higher mass brackets.

The relationship between structural mass and total mass of the target design satellites can then be expressed over the entire mass range of interest by

$$W_{str} = 0.125 \; W_{total}$$

This design factor should present no problems in realization.

The 700 lbm satellite structural mass estimate is derived from the 150 lbm structures estimate by a) reducing it proportional to the average combined radius estimates for a series of satellite dimension ratios and b) reducing it by the ratio of mass to be structurally supported. This results in a first estimate,

$$W_{str} = 150 \times \frac{2.5}{3.} \quad \times \quad \frac{700}{1200} \quad = \quad 73 \text{ lbm}$$

with a variation of about $\pm$ 5 lbm. On the other hand hub structure, cradles and minor items do not change appreciably with size; therefore the target design value of k_{str} for the 700 lbm satellite is accepted as also .125.

To obtain a corresponding value of k_{str} for a structural design which stresses the state-of-the-art, light weight materials and composites are considered. An estimate is obtained by observing the ratio of density over yield stress for competing materials: changing the satellite structure from aluminum to titanium,

$$\frac{M_1}{M_2} = \frac{\rho(Ti)/s(Ti)}{\rho(Al)/s(Al)} = \frac{.165/100}{.1/40} = \frac{.00165}{.0025} = .66$$

which would indicate

$$k_{str} = .125 \times .66 \quad = \quad 8.3\%$$

While such a structure can probably be built, we prefer to accept

$$k_{str} \quad = \quad .1$$

as a more realistic estimate of an "optimum" structure. This value is used elsewhere in this report for both the Aeronomy 1200 and 700 "optimum" designs.

5.2.3 Data System

The data system provides timing, conditioning, formatting, programming, and recording of all data from the experiment and the payload housekeeping instrumentation. The system consists of a signal conditioner, PCM encoder, recorder, and record programmer. The encoder accepts inputs from the signal conditioner and other sources and outputs either to the PM transmitter or directly to the tape recorder. The tape recorder provides storage of the data, and upon command, outputs the stored data to the FM transmitter. The function of the signal conditioner is to provide the circuitry necessary to condition transducer outputs to the proper level for entry to the encoder. The record programmer enables the experimenter to choose precisely the record time and the time between recordings.

Encoder

The encoder is required to accept analog and digital inputs, sample and convert the analog signals and format all data into a PCM wavetrain. The data rate of the encoder must be high in order that the payload spin rates may be resolved and the arc distance between samples may be kept small.

A data rate of 8, 640 bits per second was successfully utilized by the experimenter for similar experiment objectives. The data rate significantly affects the tape recorder design as it pertains to the number of channels, tape length, and reproduce speed. A 300 kHz bandwidth, compatible with STADAN, was selected, limiting the reproduce data rate to approximately

90,000 bits per second. This provides a 10:1 approximate data compression ratio with an input bit rate of 8,640 bits per second.

Tape Recorder

The tape recorder is an endless loop type capable of recording and reproducing in its original form the PCM digital information. A saturation recording technique is used.

The major characteristics required of the tape recorder are:

1) 8640 bits/second - Input data rate
2) 6 channels
3) 394 bits/tape cm./track
4) Record speed 3.66 cm./second
5) Play back speed 36.6 cm./second
6) Tape length 55. meters
7) Capacity 1.30×10^7 bits
8) Playback bit rate 86,400 bits/second

Internal timing is provided to permit termination of the reproduce mode 3.0 minutes after initiation. The actual playback time required is 2.5 minutes which provides a data overlap to insure all data is received and taped at the receiving ground station.

The recorder will consist of the following major assemblies:

1) Tape transport mechanism
2) Signal electronics
3) Motor power supply
4) Pressurized transport case
5) Interconnecting electrical cables and terminals
6) Playback timing and switching mechanism
7) Recording and playback formatter.

Required inputs from the other payload systems are:

a. 400 Hz single phase square wave for use in driving the motor, at 6 volts peak-to-peak from system clock and 100 ohm source.

b. 28 volts, DC, from the power supply at two watts maximum for record, and four watts maximum for reproduce.

c. Command pulses from control circuits at 12v, 35 milliseconds for record, reproduce, and recorder off.

d. Data input from encoder.

Record Programmer

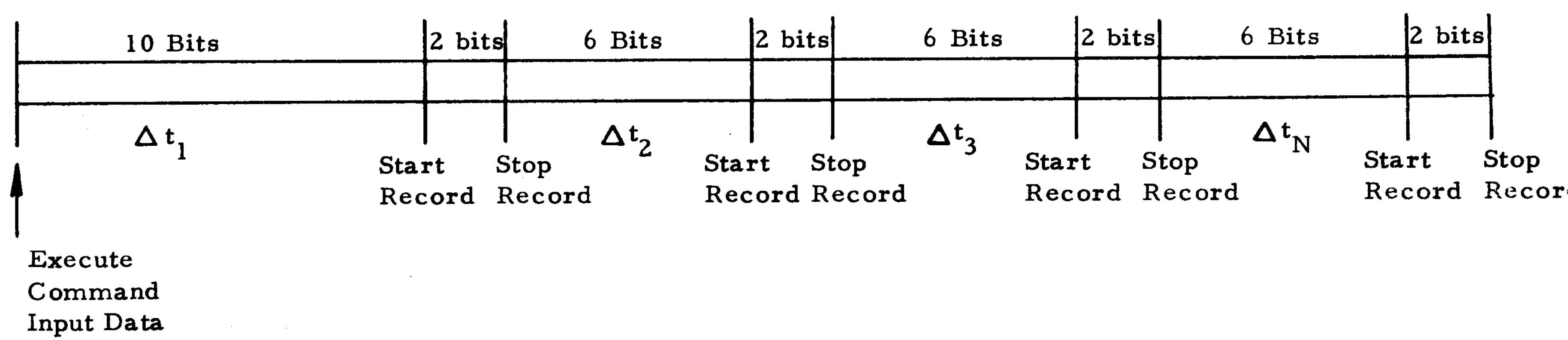
The data requirements of the experimenter are twofold:

1) A programmable sequence of data observations over various latitudes to provide approximately 2 hours of data per day and/or,

2) To make continuous data observations below 240 km (Δt = 25 minutes).

To implement the above requirements, a programmer can be designed with a format such as that of figure 5-18. The number of time increments (Δt) can conveniently be modified to any required number. A preliminary design goal of 10 increments was chosen as representative.

The time quantum was chosen as one minute per bit for each Δt. Each counter is clocked and counted down to zero. The word corresponding to the time from the command execute until the first record period, Δt_1, is 10 bits in length providing a maximum capability of 1023 minutes delay (approximately 17 hours). The remaining Δt's are 6 bits in word length providing a maximum capability of 63 minutes between record times (2/3 of an orbit).



RECORD PROGRAMMER FORMAT
FIGURE 5-18

In addition to the delayed execution commands, a capability has been incorporated to allow a selection of record durations. Two bits at the end of each delta-time register give a recording duration choice of from one to four minutes at one minute increments as follows:

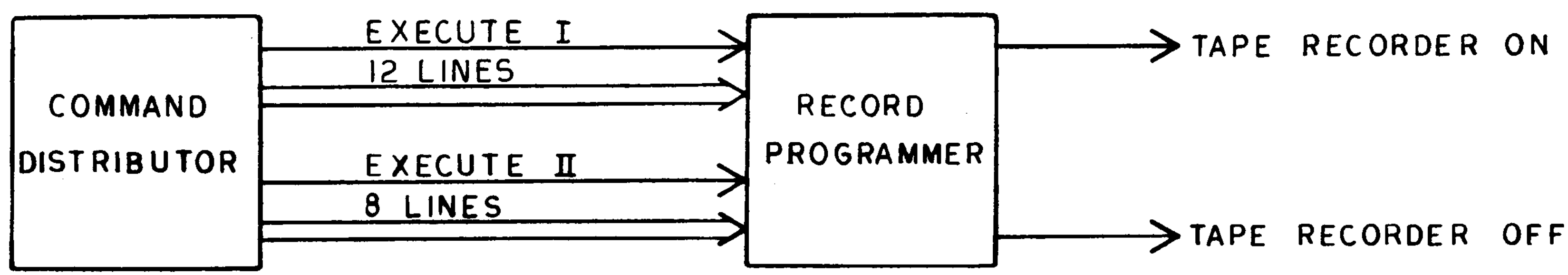
Bit Pattern	Record Time
00	1 minute
01	2 minutes
10	3 minutes
11	4 minutes

By setting all delta-time stages (except the first) to zero, up to 40 minutes continuous recording can be accomplished; however, the recorder capacity is only 25 minutes and the registers must be properly adjusted to provide only 25 minutes. This method of operation would be used to satisfy the second data requirement of continuous data collection below 240 km. The delta-time registers are prescanned so that at the end of the record time countdown the recorder stop command can be overridden when the next delta-time register is zero.

The input of data to the programmer memory can be accomplished as shown in figure 5-19. The 10 bit word corresponding to Δt_1 and 2 bits record time can be parallel shifted into the counter with "execute I". The remaining data is multiplexed into the respective counters by "execute II", since all that is required is that the "execute II" pulses be counted to enable each set of 8 bit words (Δt and a record time).

The required number of command decoder bits is 22. The time required to input all information to the programmer is 4.5 seconds.

The programmer requires a total power of 1.0 watt, and is packaged in a volume of 325 cubic centimeters.



BLOCK DIAGRAM RECORD PROGRAMMER

FIGURE 5-19

5.2.4 Communication System

The communication system is similar to that described previously. The major change is in the extended utilization of the command decoder to input timing information to the programmer. The antenna requirements are more complex for a spinning body but can be met with four fixed antennas in a turnstile arrangement.

5.2.5 Power System

The design parameters for the solar power system are established in the paragraphs following. The unit solar power derivation is based on the solar cell efficiency considerations discussed in Approach I with the exception that protective glass slides 0.006 inches (1.5 millimeters) including a ultriviolet and infrared reflective coating are used over the solar cells to reduce radiation and temperature degradation during the 12 months operation in space. A transmission loss through the glass of 8 percent and a radiation degradation of 5 percent were used to adjust the power available from the cells to 9.5 mw per square centimeter at the 50 degree celsius operating temperature. The 90 percent packaging efficiency assumption reduces the unit power to 8.6 mw/cm^2 or 86 watts per square meter of true surface area.

The nominal orbit provides the following parameters:

Apogee	800 KM
Perigee	120 KM
Orbits/day	15.37
Orbit Period	93.64 minutes
Sunlight Time	54.31 minutes
Shadow Time	39.33 minutes
Sunlight/Shadow	58 percent

For the above orbit and a knowledge that 2 hours of experiment operation is required, an energy balance equation may be written for any single day's operation as:

$$\left[(P_e \cdot 2 \text{ hours}) + N\,(T-t)\,P_{s/c}\right] \quad \frac{1}{E_f} \quad + NP_{s/c} \; t = Nt \; P_p \qquad (1)$$

E_f	=	Battery efficiency
P_p	=	$P_o A_o$ = Power from solar panels
$P_{s/c}$	=	Power required by spacecraft
P_e	=	Peak power required by experiment
N	=	Total orbits/day
T	=	Orbit period
t	=	Sunlight portion orbit period

The parameter P_p can be expressed as the unit power from the solar panels (P_o) and the projected area (A_o). The solution to the above equation for A_o is shown in figure 5-20 as a function of the average power supplied to the spacecraft. The total solar array projected area for an earth oriented body (local vertical), a disc and a sphere is shown in figure 5-20 . To provide 15 watts average power to the spacecraft requires a minimum projected area of 0.418 square meters. The corresponding total solar array required for a cylinder is 2.0 square meters, and for a sphere is 1.59 square meters. An earth oriented body requires an array of 1.81 square meters since the inclination of the orbit plane to the sun line must be compensated. An assumed worst case inclination of 61.5^{o} was utilized for the data.

It is noted from figure 5-20 that the sphere requires the minimum total solar array for a given power requirement. The drag area of the sphere is the area of a circle whose radius equals that of the sphere.

FEASIBILITY STUDY OF AN AERONOMY SATELLITE

FINAL REPORT

TO

THE NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

GODDARD SPACE FLIGHT CENTER

UNDER CONTRACT NO. NAS5-9346

5 AUGUST 1966

SPACE CRAFT, INC.
8620 SOUTH MEMORIAL PARKWAY
HUNTSVILLE, ALABAMA

PREFACE

This is a final report submitted to Goddard Space Flight Center under Contract NAS5-9346 entitled "Study of An Aeronomy Satellite". Two design approaches are presented for using propulsion to extend the satellite lifetime in low perigee orbits. Parametric lifetime tables are included.

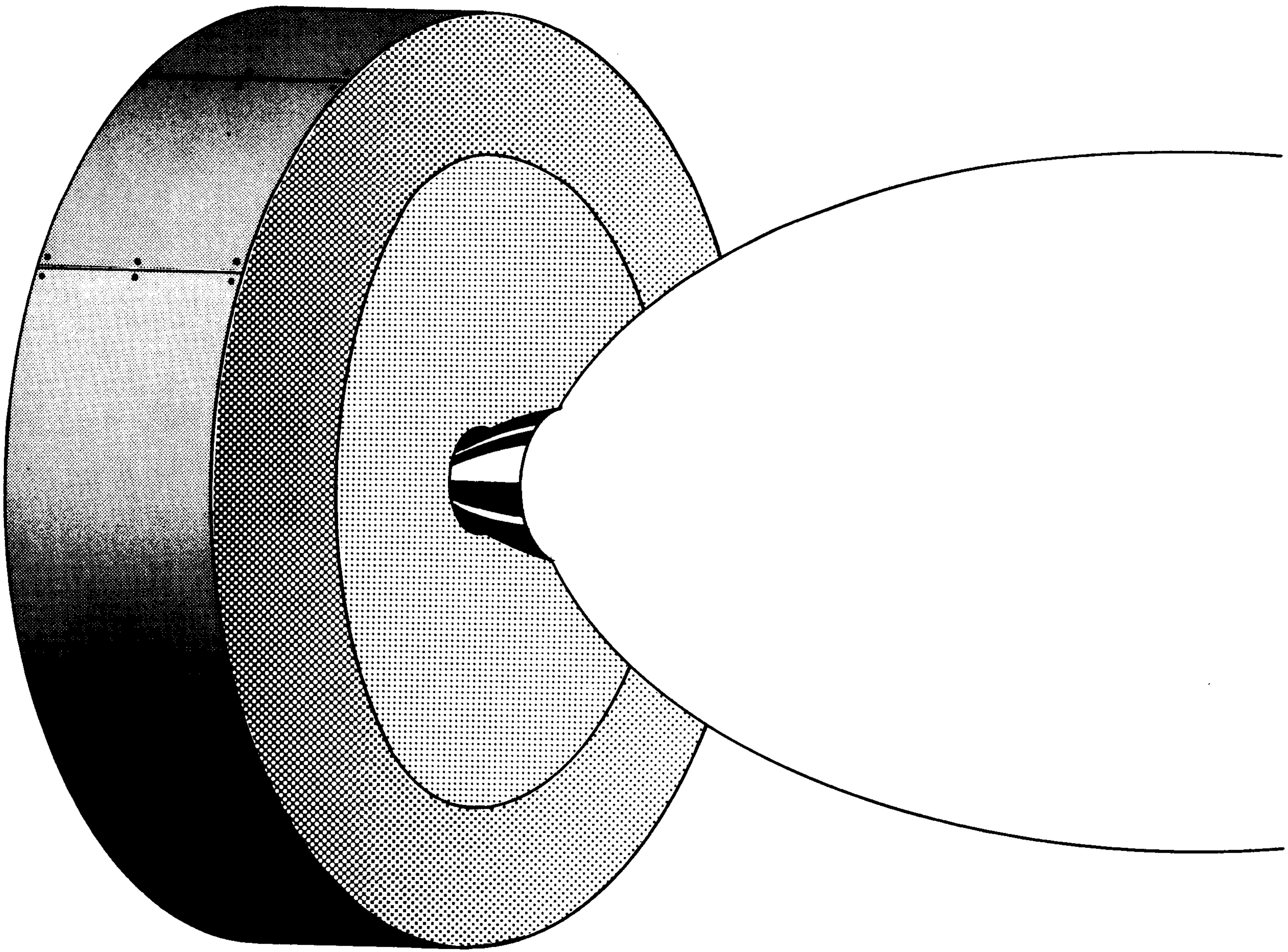
The contract was administered for GSFC by Mr. H. W. Spencer as Technical Officer and Mr. D. Grimes as project engineer.

TABLE OF CONTENTS

TABLE OF CONTENTS (CONT'D)

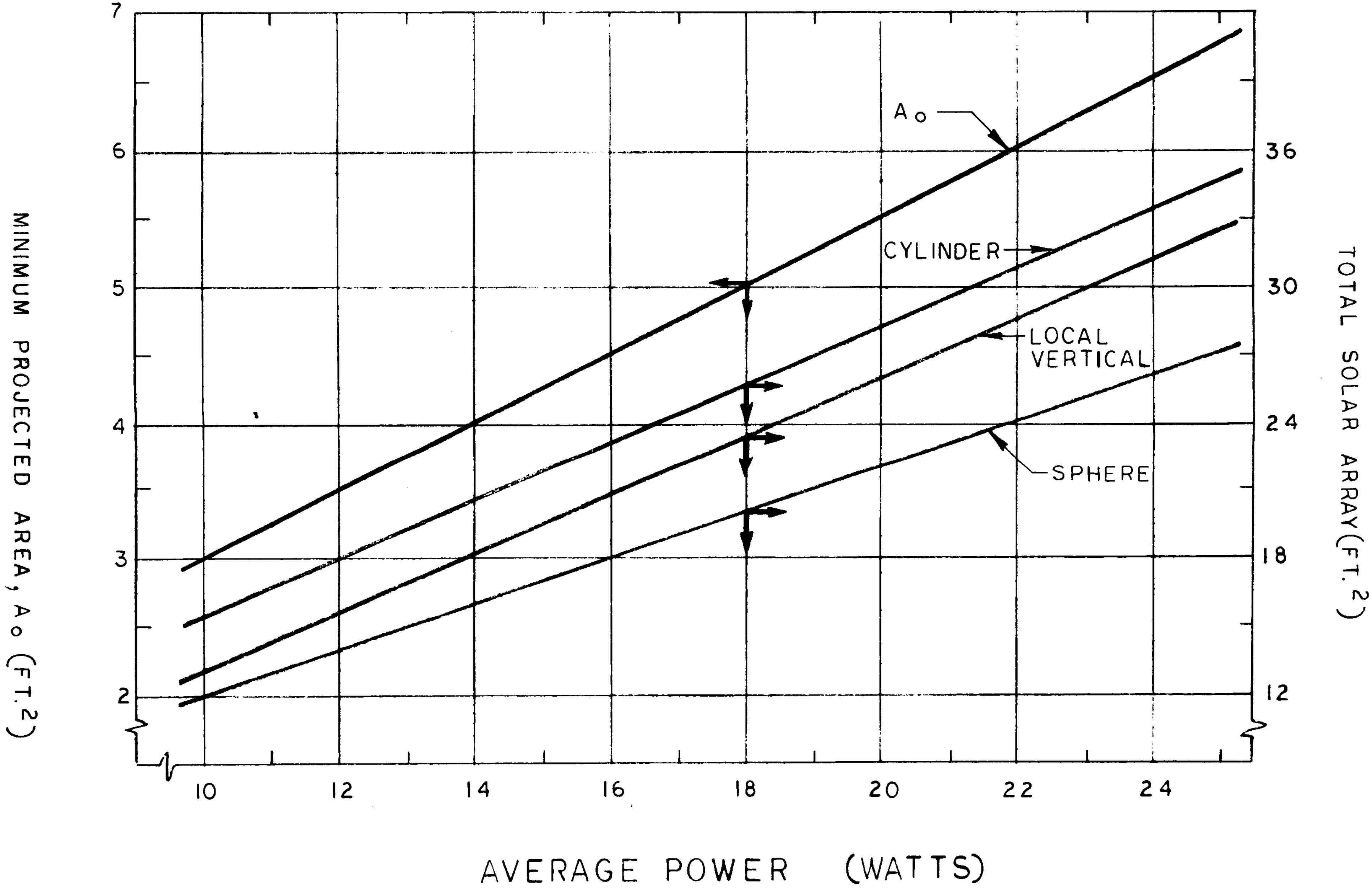
LIST OF FIGURES

LIST OF TABLES



ARTIST CONCEPTION, AERONOMY/SS SATELLITE

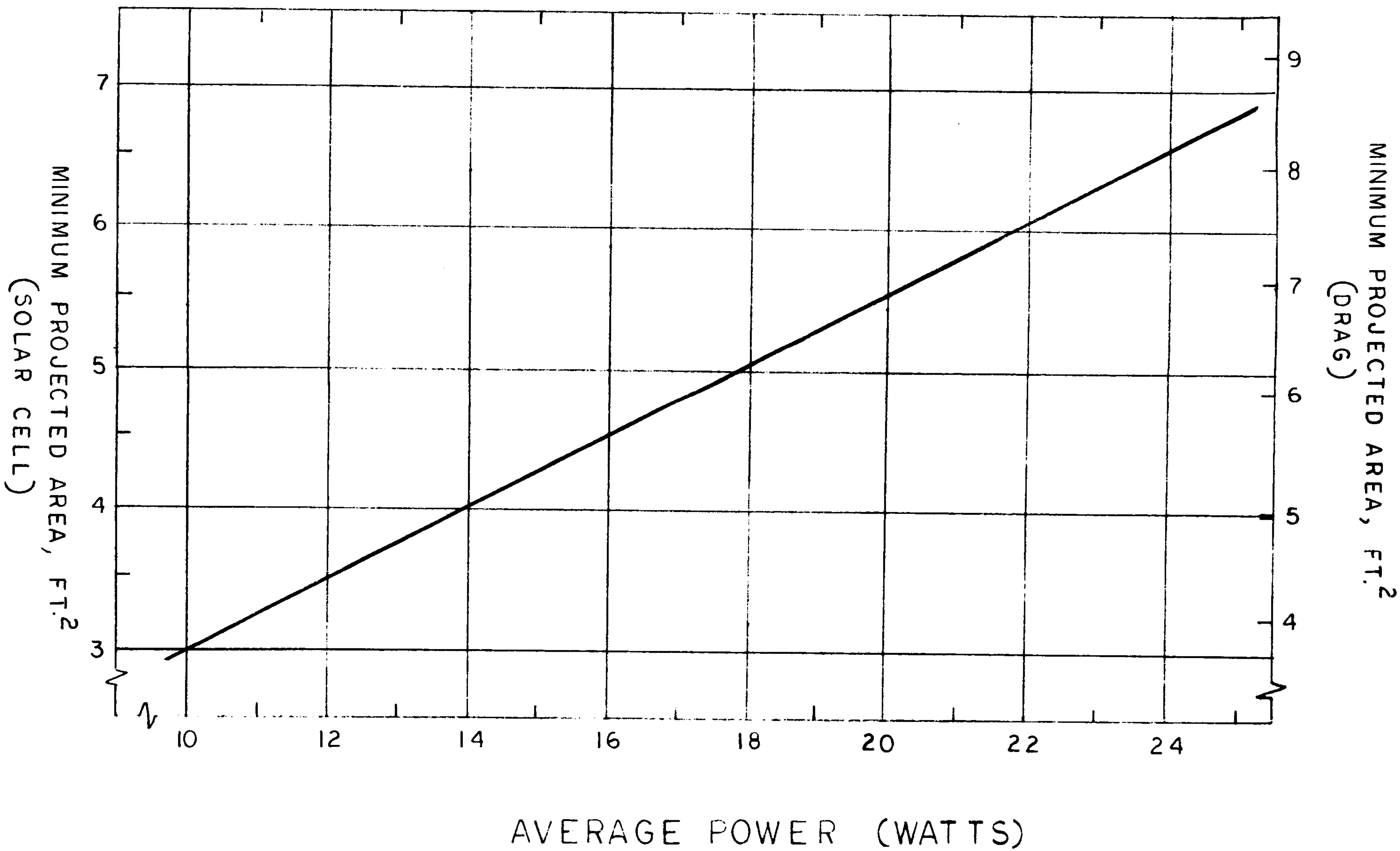
FIGURE 1-1



PROJECTED AREA AND TOTAL SOLAR ARRAY VS. AVERAGE POWER
FIGURE 5-20

The required solar panel area for the cylinder is determined taking the worst case condition, that is, when the orbit plane is in the earth-sun line. This condition provides a projected area of $A_o = 2\ rh$, where r is the radius and h is the height of the cylinder. The total surface area of the periphery of the cylinder is then $2\pi rh$. An equal projected area, A_o, is required on the top and bottom of the cylinder to provide the minimum power in any attitude. Therefore, the total solar array area is $\pi A_o + 2 A_o$ or $A_o (\pi + 2)$. The frontal area, or drag area, is directly related to the minimum required projected solar cell area by the coverage factor. Figure 5-21 shows this relationship for a cylinder assuming an 80 percent area coverage for the solar cells. The lifetime of the spacecraft is directly a function of the drag area, and the power available is proportional to the drag area. Therefore, to fix either the required drag area or the power defines the other variable. A compromise must be made since the minimum drag area is also a minimum power available.

To provide a comprehensive comparison of the cylinder and the sphere, bodies of equal volume were investigated. The physical constraint of the protective shroud limits one dimension of the body to a maximum of 5 feet (1.5 meters). The minimum height of the cylinder was chosen as 1 foot (30.5 centimeters) to allow realistic propellant tankage dimensions. It can be noted from figure 5-22 that the minimum projected area ratio (cylinder/sphere) is significant in terms of drag effectiveness. The cylinder of the most favorable area ratio will provide a projected solar array of 3.4 to 4.6 square feet (0.32 to 0.43 square meters) corresponding to power levels of 11.5 to 16.3 watts to the system. The respective drag area is between 4.25 and 5.75 square feet (0.40 and 0.53 square meters). The projected area for the sphere of corresponding volume is 7.07 to 9.6 square feet (0.66 to 0.89 square meters). The power available is between 20 and 28 watts.



SOLAR CELL AREA AND DRAG AREA VS. AVERAGE POWER
FIGURE 5-21

		SPHERE				CYLINDER								MINIMUM PROJECTED AREA RATIO
$V_d = V_s$ (Volume)		r_s, radius		A_s Area		*r_d, radius		* h, height		A_d Area		A Solar Array		A_d/A_s
(m^3)	(ft^3)	(m)	(ft)	(m^2)	(ft^2)	(m)	(ft)	(m)	(ft)	(m^2)	(ft^2)	0.8 A_d (m^2)	(ft^2)	
0.952	33.6	0.610	2.0	1.17	12.6	0.762	2.5	0.518	1.7	0.790	8.5	0.632	6.8	0.67
0.637	22.5	0.533	1.75	0.892	9.6	0.762	2.5	0.351	1.15	0.534	5.75	0.427	4.6	0.59
0.399	14.1	0.457	1.5	0.657	7.07	0.646	2.12	0.305	1.0	0.394	4.24	0.316	3.4	0.60
0.231	8.17	0.381	1.25	0.455	4.9	0.491	1.61	0.305	1.0	0.299	3.22	0.242	2.6	0.65
0.118	4.18	0.305	1.0	0.292	3.14	0.351	1.15	0.305	1.0	0.214	2.3	0.167	1.8	0.73

Volume

$V_d = \pi r_d^2 h$

$V_s = 4/3 \pi r_s^3$

Projected Area

$A_d = 2 r_d h$

$A_s = \pi r_s^2$

* Constraints: 1) $r_d \leq$ (2.5 ft)

$h \geq$ (1.0 ft)

DRAG EFFICIENCY COMPARED TO EQUAL VOLUME SPHERE

FIGURE 5-22

Thus, the cylinder provides a reduction in drag area over a sphere of equal volume but it does so at a sacrifice in available power.

Battery

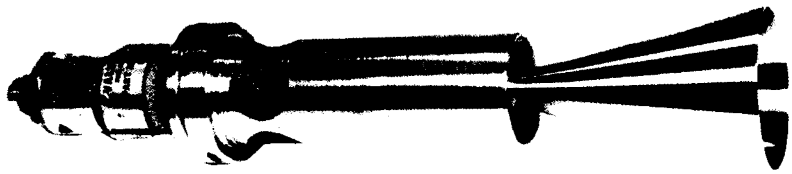
The battery requirements for the spacecraft are determined by the peak experiment energy requirements. The peak energy requirement occurs when recording through a perigee pass. The associated time is approximately 25 minutes operation at 130 watts or 55 watt-hours. For a 10 percent depth of discharge, the required battery capacity is 550 watt-hours or approximately 20 amp-hours at 28 volts. The number of charge and discharge cycles per year assuming a maximum of 4 operational perigee passes per day is approximately 1500 cycles.

The nickel cadmium battery is recommended for this application due to its large number of charge and discharge cycles (greater than 20,000 cycles at 10 percent depth of discharge) and the favorable charge characteristics. The maximum overcharge capability of a nickel-cadmium battery is c/10, or 2 amps for a 20 A-H battery which somewhat simplifies the battery charge regulator. The 20 AH nickel cadmium battery pack would weigh a maximum of 23 kilograms.

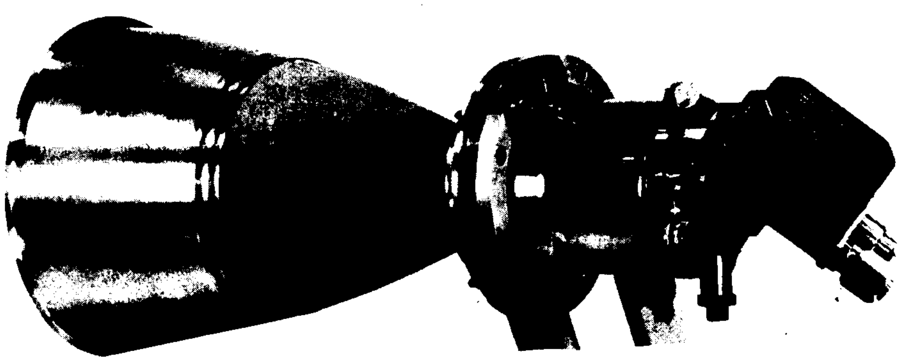
5.2.6 Propulsion

The Aeronomy/SS basic mission maneuver as described in section 4.1.1 requires 75 individual thrust periods. A number of rocket engines can meet these requirements: discussion is limited to hydrazine decomposition engines using Shell-405 catalyst, and to the C-1 bypropellant engine using hypergolic monomethyl hydrazine (MMH) and nitrogen tetroxide (N_2O_4). Figure 5-23 and 5-24 show the C-1 engine and its principle characteristics.

TYPICAL MONOPROPELLANT ENGINE



C-1 BIPROPELLANT ENGINE



TYPICAL ENGINES

FIGURE 5-23

PARAMETER		SPECIFICATION	DEMONSTRATED
1.	Thrust	85 - 100 lb.	74 - 110 lb.
2.	Chamber Pressure	79 - 100 psia	75 - 120 psia
3.1	Nominal Specific Impulse	272 sec. @ E = 8.0 307 sec. @ E = 60.0	
3.2	Minimum Specific Impulse	266 sec. @ E = 8.25 301 sec. @ E = 60.0	265 - 269 sec. @ E = 8.25 304 sec. @ E = 60.0
4.0	Propellant Combinations	N_2O_4/MMH N_2O_4/50% UDMH + 50% Hydrazine	yes yes
5.0	Mixture Ratio	1.6	1.3 - 1.9
6.0	Supply Pressure, Dynamic	178 (Min.), 310 (Max.), psia	yes
7.0	Supply Temperature	+20°F to +130°F (N_2O_4/MMH) +40°F to +100°F (N_2O_4/50-50)	yes yes
8.0	Minimum Impulse Bit	0.4 $\pm$ 0.2 lb. sec.	0.4 $\pm$ 0.1 lb. sec.
9.0	Thrust Rise Time (Biprop. valve)	0.020 sec. to 90% rated thrust	0.021 sec. to 90% rated thrust
10.0	Thrust Delay Time (Biprop. valve)	0.030 sec. to 10% rated thrust	0.023 sec. to 10% rated thrust
11.0	Firing Duration - Ablative Radiation	755 sec. (Target - 1800 sec.) 2000 sec. (Target - 5000 sec.)	2733 sec. 2900 sec.
12.0	Basic Engine Firing Duration	5000 sec. Target	5166 sec.
13.0	Cycle Life (No. of Starts)	30,000	Now Being Demonstrated
14.0	Characteristic Velocity	92.0 to 94.16%	94.5% (Average)
15.0	Maximum Skin Temperature (Buried Application)	600°F	Now Being Demonstrated
16.0	Basic Engine Weight (Biprop.)	5.0 lb.	Now Being Demonstrated
17.0	Engine Weight (Biprop.) with Radiation Skirt	6.26 lb.	Now Being Demonstrated
18.0	Nominal Operating Voltages for Biprop. valve engine	28 VDC	yes
19.0	Demonstrated Reliability at end of Qualification Program	0.99 @ 50% Confidence Level	

PRINCIPAL CHARACTERISTICS, C-1 ENGINE

FIGURE 5-24

In favor of the C-1 engine is a major NASA funded development program which is effectively on schedule and performance mileposts. Secondly, the requirement for man-rating will provide high confidence factors for reliable space operation. The thrust level of 100 lbf nominal, and the very low combined engine, bell and valve mass make the engine suitable for the Aeronomy application. A projected low cost is less important, but also significant.

Opposed to use of bipropellants are the slightly greater complexity due to dual supply and valving requirements; the potential for hypergolic ignition of leaking propellants; and the tank sizing problem. The C-1 engine operates at a volume ratio of about 1.2 oxidizer/fuel, and configuration characteristics for attaining this are more limiting than for the monopropellant designs.

The immediate advantage to this study of a bipropellant system based on the C-1 engine are:

> A volume specific impulse of 22,200 lb-sec/ft^3 (opposed to a hydrazine value of 14,100 lb-sec/ft^3)
>
> As a result, a significantly higher satellite energy content.

The advent of a reliable hydrazine decomposition catalyst, shell-405, permits repeated restarts without use of the formerly necessary hypergolic starting slugs. As a result a very simple monopropellant system can be designed. Kidde offers a 250 lbf engine in advanced development which appears suitable. Although specific impulse is at the 225 lbf-sec/lb level, the higher thrust will improve the system efficiency somewhat, and heat transfer problems are lessened. Slight degradation of catalyst efficiency with operating life must be accounted for in design.

The immediate advantages to this study of a monopropellant system based on Shell-405 are:

extreme simplicity in execution,

freedom in packaging the propellant.

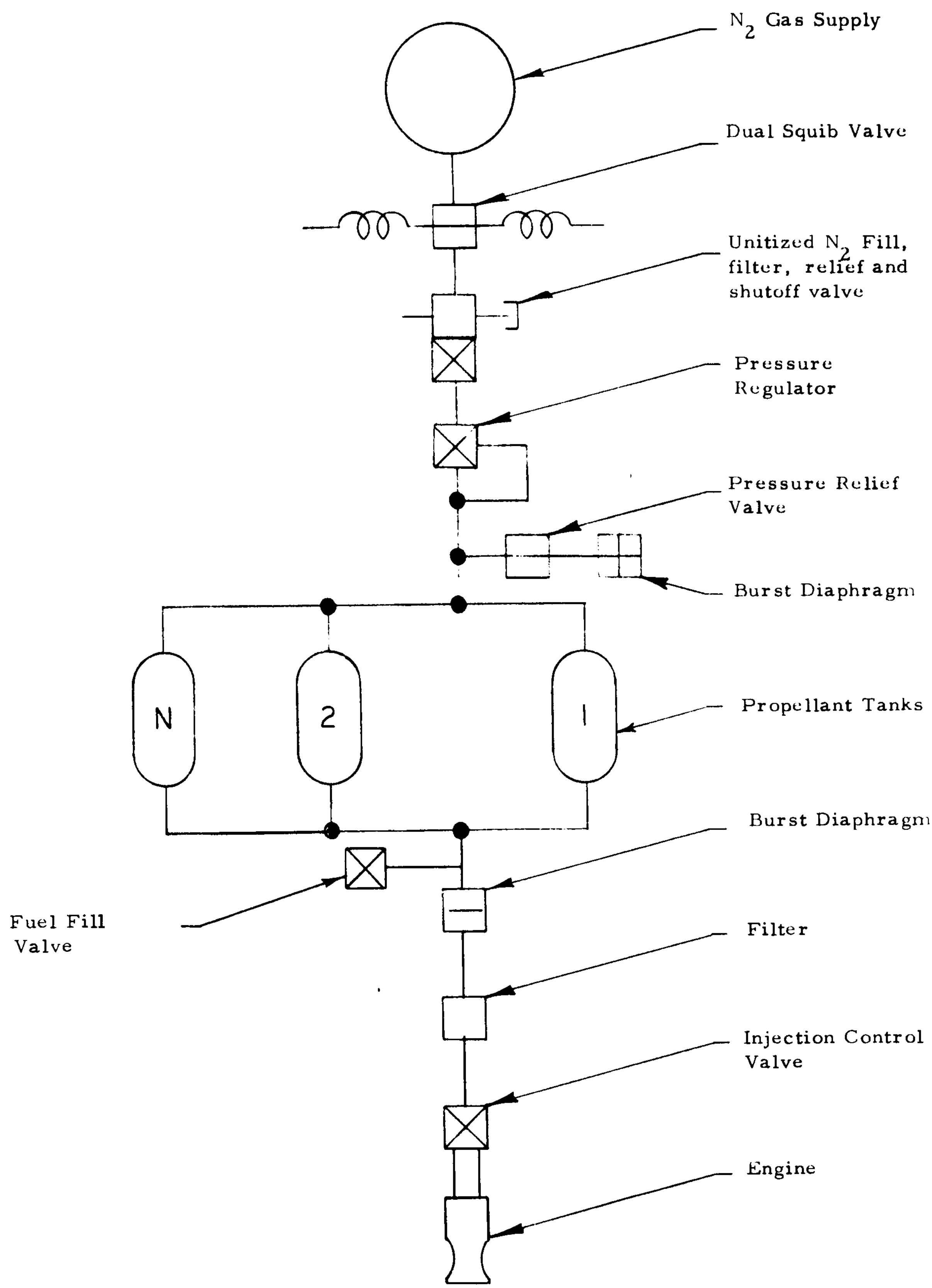
Figure 5-25 and 5-26 show a schematic of both types of propulsion systems.

Propellant Tanks

The tanks represent over 1/3 of the satellite's entire inert mass, and are the largest contributing item. They will operate at about 250 psia. Figure 5-27 shows small flight hardware tank factors. The ordinate is in lbm of tank per cubic foot of volume; the abscissa is in gamma units, g = 1/r as shown. The upper band shows three designs at .02 lbm/in^3; the lower band data varies from .095 to .0135 lbm/in^3. Note that the effect on structural mass is slight with gamma. The band on the ordinate between .011 and .014 represents 5000 psi conventional pressure spheres, with the lower limit of Titanium and the upper of 4130 steel.

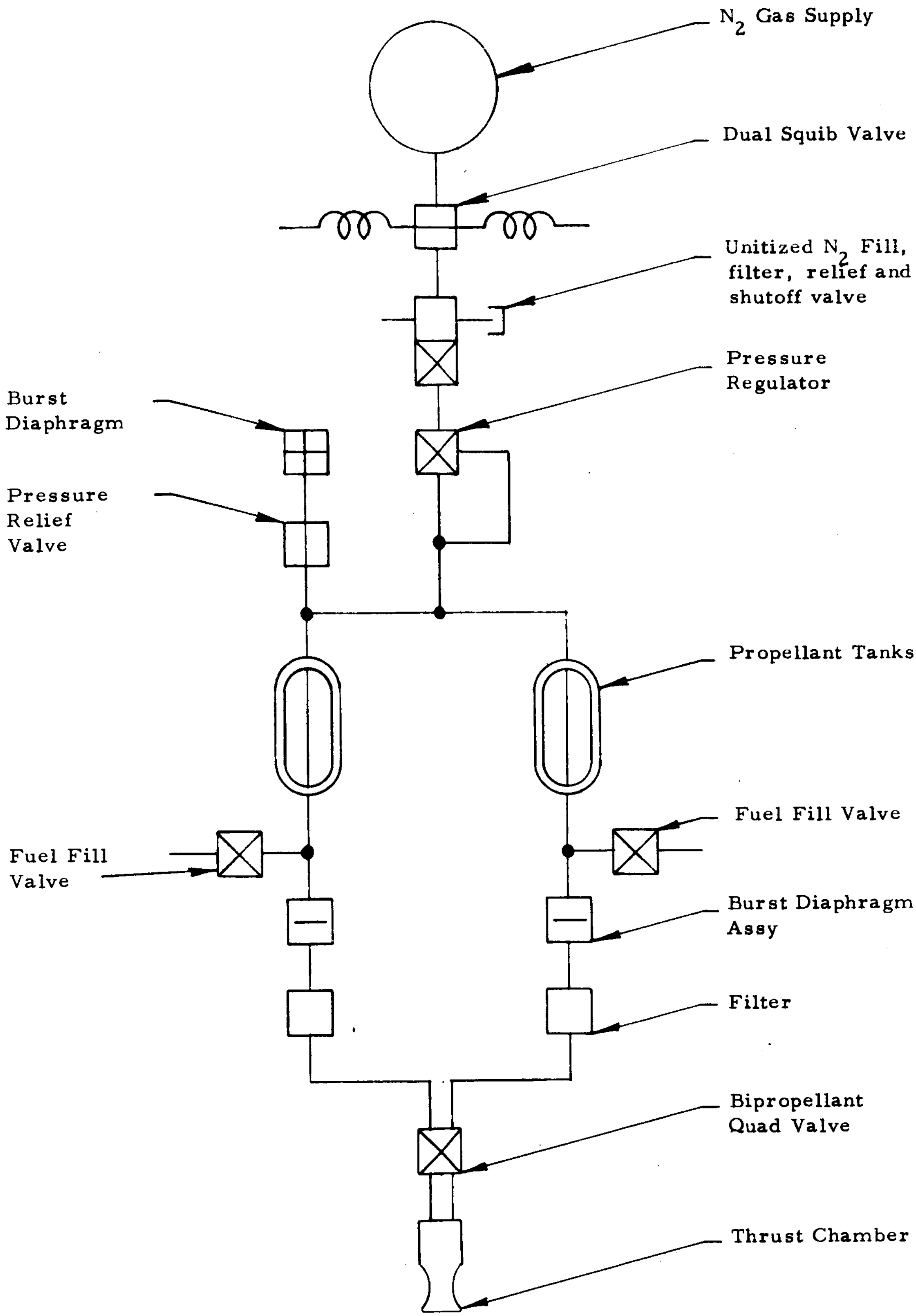
The tank gamma for designs in the Aeronomy 1200 region will lie in the crosshatched region of the figure. The heavy lines indicate our choice of a conservative .0125 lbm/in^3 for the target design, and .0093 lbm/in^3 for the optimistic design. Both of these values can be improved, we believe, once development gets under way. The values are 21.6 lbm/ft^3 and 16.1 lbm/ft^3, and values to 13 lbm/ft^3 should be achievable. The stacking of margins and safety factors must be examined critically, since savings in tank weight are directly translatable into increased satellite mission life. These margins and factors should therefore be stated to satisfy unmanned mission requirements only.

By introducing the classic hoop stress formula into the sphere surface via the wall thickness parameter, it is shown that tank mass is independent of tank volume.



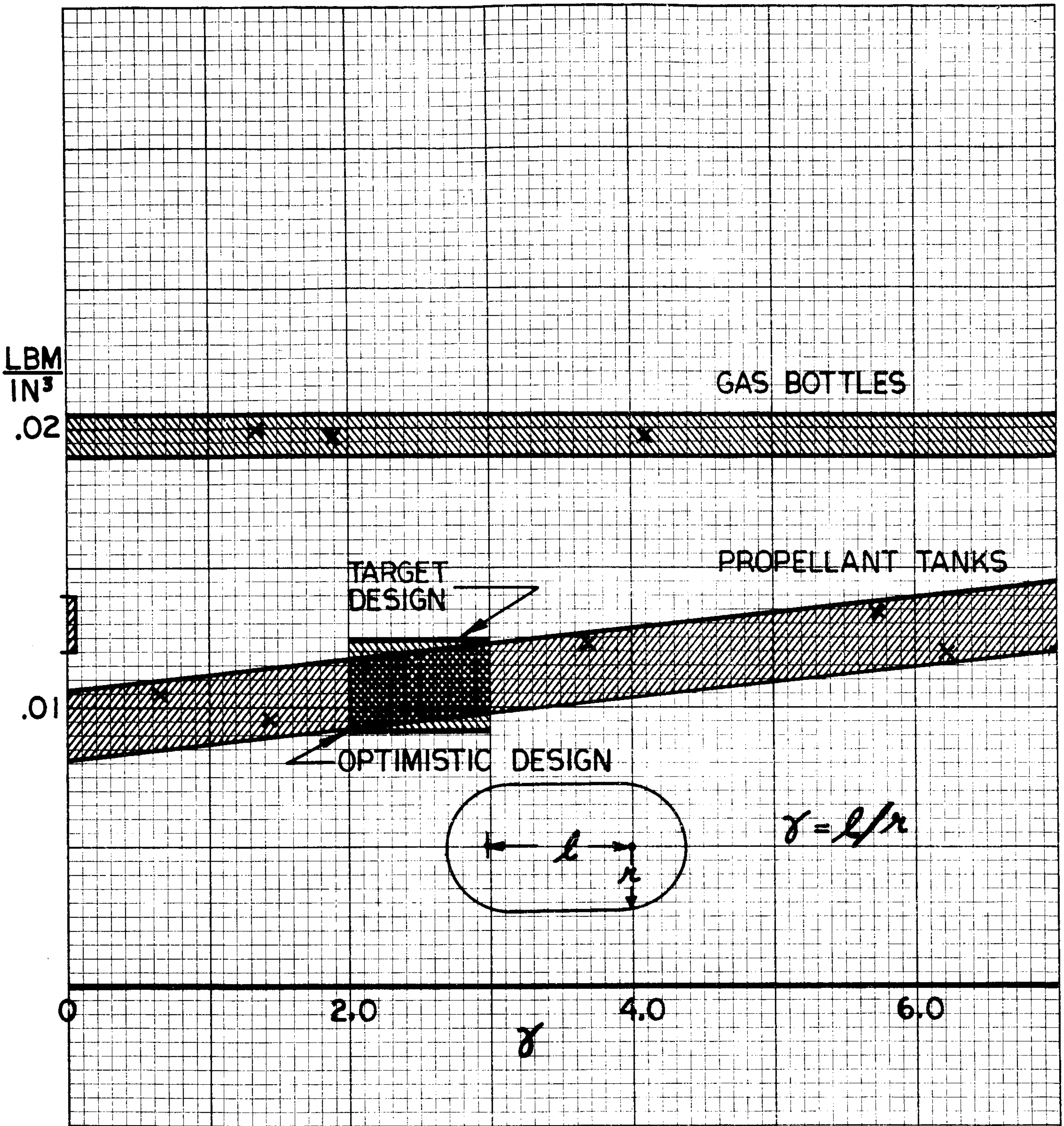
PRELIMINARY SCHEMATIC, HYDRAZINE MONOPROPELLANT PROPULSION SYSTEM

FIGURE 5-25



PRELIMINARY SCHEMATIC BIPROPELLANT PROPULSION SYSTEM

FIGURE 5-26



PROPELLANT MASS TANK PARAMETERS

FIGURE 5-27

Figure 5-27 showed this to hold even for substantially elongated tanks. Glass wrapped aluminum permits non-isotropic material properties to be exploited so that shape factors are far less important, and spheres are not the ultimate answer anymore. Heavy wall vessels of passivated aluminum, however, do reduce the propellant fraction carried per unit of total mass. The technology in this area should be reviewed for overall mass fraction improvement potential.

Propellant and Tank Mass

Using eqn (1) of Appendix A

(1) $$W_t = W_{str} + W_{ta} + (W_{pp} + W_{ACS}) + (W_{ex} + W_{ss})$$

Define: W_t = total initial satellite mass

$W_p = W_{pp} + W_{ACS}$, the total propellant mass

$W_{ta} \doteq$ tank mass

$W_f = W_{ex} + W_{ss}$, all other masses.

$W_{str} = K_{str} W_t$, see section 5.2.2.

(2) Then $$W_t = K_{str} W_t + W_{ta} + W_p + W_f.$$

Since a relationship with selected system constants is desirable, and also a relation to satellite dimension, let

(3) $$V_p = .97 \quad 1 - \frac{D}{D'} \quad V_{geo}$$

where: V_p = net propellant volume

V_{geo} = computed volume based on external dimensions

$.97$ = ullage factor

D = virtual tank density, taken from preceding section

D' = tank material density

(4) Then $$W_p = \rho \, V_p$$

(5) and $$W_{ta} = D \, W_p$$

where ρ = propellant density

Using (3), (4) and (5), equation (2) can be rearranged to the desired forms:

(6) $$W_t(V_p) = \frac{(.97 \quad +D) \quad (1 - \frac{D}{D'})}{1 - K_{str}} \quad V_{geo} \quad + \quad \frac{W_f}{1 - K_{str}}$$

(7) $$W_t(W_p) \quad = \quad \frac{(1 + \frac{D}{.97})}{1 - K_{str}} \quad W_p \quad + \quad \frac{W_f}{1 - K_{str}}$$

Equations (6) and (7) are plotted in Section 3, figures 3-1 thru 3-4, for the following choice of constaints:

	Monopropellant					Bipropellant				
Satellite	ρ	D	D^1	k_{str}	W_f	ρ	D	D_1	k_{str}	W_f
Optimum 1200	62.4	16.1	180	0.1	209	74.8	16.1	180	0.1	2[illegible]
Target 1200	62.4	21.6	180	0.125	243	74.8	21.6	180	0.125	2[illegible]
Target 700	62.4	21.6	180	0.125	201	74.8	21.6	180	0.125	2[illegible]

Equations (5) and (6) give total satellite mass as linear functions of the geometric tank volume and the propellant mass respectively. The functions are graphed in section 3.0, and serve to relate satellite geometry to the various mass parameters via the tables of Appendix B.

Summary of Propulsion System

It is found that both the C-1 bipropellant engine and the hydrazine-catalytic decomposition engine are suitable for the Aeronomy/SS satellite mission. The propulsion system design presents no unusual problems and the performance calculations of this report can be met. It is recommended that further studies centering on practical tank system questions be executed to explore mass reduction possibilities.

5.2.7 Altitude and Reaction Control System

Basic Reference System

The basic reference system for the spin stabilized approach is an orbit fixed coordinate system. The spin axis is to be maintained normal to the orbit plane. The orbit plane rotates in inertial space due to oblateness effects of the earth. The spin axis tends to remain fixed in space and thus the direction of the spin axis must be changed to keep it aligned with the normal to the orbit plane. This correction will in general be on the order of a few degrees per day.

During propulsive maneuvers the spin axis of the satellite must be approximately aligned with the orbital velocity vector. Thus, the attitude during propulsives phases is 90° from the reference cruise attitude.

Attitude Control System

The attitude control system must provide two basic functions. It must be capable of making the spin axis track the normal to the orbit plane during cruise phases; and, it must be capable of slewing the spin axis to the proper attitude for propulsion maneuvers. Attitude reference can be accomplished by use of solar aspect sensors, earth aspect sensors, and antenna polarization measurements. Measurements from these sensors can be fed into a ground computation loop and the appropriate corrections or attitude changes may then be relayed to the satellite through a command link.

The attitude changes are then effected by a reaction control system which precesses the spin vector to the proper direction. Any residual motion of the spin vector about the angular momentum vector must then be damped out. The spin axis of the satellite will be designed to be the axis of maximum inertia so that energy dissipative type damping may be employed.

Reaction Control System

Two types of reaction control systems are particularly attractive for use in the control of spin stabilized satellites. These are magnetic torqueing and "two pulse" mass expulsion type systems.

Magnetic torqueing is accomplished by generating a magnetic moment between the satellite and the earth magnetic field lines. This is done either by energizing a coil in the spacecraft and apply continuous power to maintain the magnetic moment, or by magnetizing a permeable rod to some high residual value and then demagnetizing it at the appropriate time. Both methods require that the angle between the spin vector and the earth's magnetic field lines be in the proper direction. It has been found by previous studies that during about one sixth of an orbital period a favorable relationship for torqueing exist. The torque generated is, in general, fixed in inertial space and residual motion of the spin axis about the satellite angular momentum vector results. Some type of damper must then be employed to remove the residual motion.

Preliminary sizing of a magnetic torqueing system may be accomplished by assuming a homogeneous mass distribution of the satellite for purposes of calculations. For a 1200 lbm satellite with a 2.5 ft radius the moment of inertia is about 100 slug-ft.^{2} The precession rate of the spin vector is given by

$$\dot{\theta} = \frac{T}{I\omega}$$

where $\dot{\theta}$ = precession rate

T = torque

I = moment of inertia

and ω = spin rate

Using a dipole representation of the earths magnetic field, assuming an electro magnet with aluminum wire and a 5 ft. diameter coil, the maximum torque that can be generated at 300 n. mile altitude is

$$T_{max} = 5.915 \times 10^{-4} \sqrt{PW}$$

where T = torque in ft. lbf.

P = electrical power in watts

and W = the weight of the coil in lbm.

With a spin rate of 10 rpm the precession is

$$\dot{\theta} = 5.915 \times 10^{-2} \sqrt{PW}$$

where $\dot{\theta}$ is the precession rate in degrees per minute. Assuming the maximum power available is about 100 watts a coil weight of 100 lbm. would provide a precession rate of about 6°/min. maximum; or an average of about 3°/min for a 90° torqueing maneuver. Thus the system could provide a 90° turn in 30 minutes. Assuming a 15 minute period of favorable torqueing per orbit two orbits would be required for a 90° turn. Note that all of the assumptions made are in our favor which of course in general is not true.

The "two pulse" mass expulsion control system is characterized by two mass expulsions jets placed some α degrees apart as shown in figure 5-28. When an attitude change is required, as determined by ground based computations; jet A is pulsed at the appropriate position in inertial space. This results in a change in position of the angular momentum vector by an amount

$$\Delta\theta = \frac{Fd\,\Delta t}{I}$$

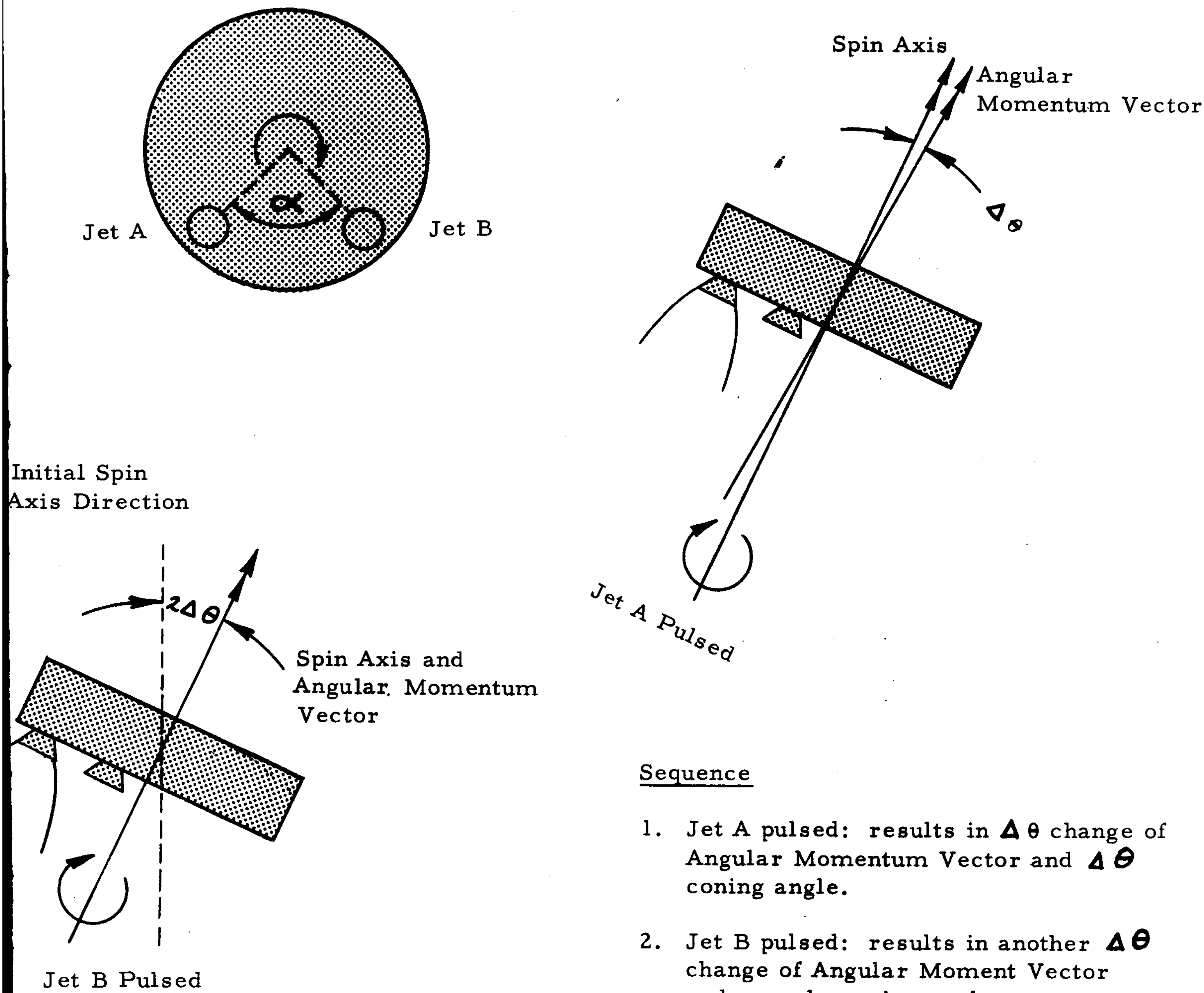
where F = average thrust level

d = moment arm of the jet

t = effective pulse width

and $\Delta\theta$ = angular change.

A coning of the spin axis about the angular momentum vector with a cone half angle of $\Delta\theta$ is also established. When jet B occupies the same position in inertial space as jet A did when it was fired, jet B is fired. This results in another $\Delta\theta$ change in the position of the angular momentum vector in the desired inertial direction. The angle α between jet A and jet B is chosen such that when jet B is fired, the coning angle established



<u>Sequence</u>

1. Jet A pulsed: results in $\Delta\theta$ change of Angular Momentum Vector and $\Delta\theta$ coning angle.

2. Jet B pulsed: results in another $\Delta\theta$ change of Angular Moment Vector and cancels coning angle.

3. Net change is $2\Delta\theta$ in attitude per revolution.

TWO PULSE REACTION CONTROL SYSTEM
FIGURE 5-28

by jet A is cancelled. The net change in attitude is then 2 $\Delta\theta$ per revolution of the satellite. Assuming a .2 sec. pulse duration, a 2.5 ft. momentum arm, a 100 slug-ft^2 inertia, a spin rate of 10 rpm and an engine thrust of about 5 lbf.; $\Delta\theta$ per pulse is about 1°, or 2° per revolution of the satellite. A 90° turn could then be made in 45 revolution or about 4.5 minutes. By changing combinations of spin rate and jet thrust this time may be lengthened or shortened. This particular type reaction control system was assumed to be employed on the spin stabilized vehicles which were evaluated for mission performance in section 4. It was found that about 55 lbm of hydrazine propellant, derated to a specific impulse of 150, is required to complete the basic mission for the 1200 lbm. configurations. The total reaction control system weight for this approach, including propellant, would be about 65 lbm. It is assumed that the propellant for the RCS is obtained from the main propulsion system propellant tanks.

It should be noted that two jets are not necessarily required for the system. One jet alone may be employed at the sacrifice of a longer time being required for a turn. The two jet approach simplifies the system somewhat and also provides a degree of redundancy.

Both magnetic and mass explusion type reaction control systems have been successfully employed on satellites. The magnetic system has been used on Explorers and the two pulse system has been employed on the Syncom series of satellites.

On the basis of this cursory analysis it appears that the "two pulse" type system is to be favored. The overall required weight is lighter, the power consumption is nil, and the time required to execute a given turn is about an order of magnitude shorter. The time for completion of a turn is important because in a thrusting attitude the vehicle is in a high drag configuration and aerodynamic energy dissipation is increased.

Spin Control

Some type of spin control device must be employed to maintain the desired spin rate. The spin rate will tend to decay due to eddy current interaction with the earths magnetic field. Conversely, the spin rate will tend to increase due to a reduction in moment of inertia caused by the expendature of propellant. The angular momentum removed from the spinning vehicle by the expanding gases is not sufficient to counteract the inertia change, and the vehicle will therefore increase in spin rate to conserve angular momentum.

To what extent spin rate increase and spin rate decay offset each other has not been determined. It appears evident, however that some type of spin control will probably be required. Either magnetic or mass expulsion type control is feasible and the choice will probably be based on the system chosen for the main attitude control functions.

-6-

PRELIMINARY SYSTEM SPECIFICATION, PART 1

FOR

A PROPELLED AERONOMY SATELLITE SYSTEM

SPACE CRAFT INC.

MODEL ____________ DATE July 22, 1966

TITLE

SYSTEM SPECIFICATION

SS-AE0001

PERFORMANCE AND DESIGN REQUIREMENTS

FOR THE

AERONOMY EXPERIMENTS SATELLITE SYSTEM

APPROVAL	TITLE	DATE
	Dir. Sys. Dev. Dept.	
	Ch. Sys. Analysis	
	Ch. Sys. Integration	

SCI FORM 810-031

1. SCOPE

This specification defines the performance, design requirements, and guidelines for an Aeronomy Satellite system to be flown on an improved Thor Delta launch vehicle. Development and test requirements are specified; all equipment and secondary specifications shall be compatible with requirements delineated herein.

2. APPLICABLE DOCUMENTS

The following documents of the issue shown form a part of this specification to the extent specified herein. In the event of conflict between the referenced documents and this specification, this specification shall be considered a superceding requirement. Additional workmanship, quality, and detail references shall be delineated in subsystem and equipment specifications.

MILITARY

MIL-E-6051C	Electrical-Electronic Equipment System Compatibility and Interference Control Requirements for Aeronautical Weapon Systems, Associated Subsystems, and Aircraft.
MIL-STD-810A	Environmental Test Methods for Aerospace and Ground Equipment.
MIL-I-6181D	Interference Control Requirement Aircraft Equipment.
MIL-STD-129C	Marking for shipment and storage.
MIL-STD-130	Identification marking for U.S. Military property.

MIL-D-70327	Drawings, Engineering and Associated Lists.
MS-33586	Definition of dissimilar metals.
MS-33540	Safety wiring, general practices.

FEDERAL

NPC 200-3	Inspection System Provisions for suppliers of space material, parts, components, and services.
MSFC-SPEC-250	Protective finishes for space vehicles, structures, and associated flight equipment, specification for.
MSFC-PROC-256	Electrical fabrication, procedure for.

3. REQUIREMENTS

3.1 PERFORMANCE

The Aeronomy Satellite is intended as a primary payload system carrying scientific measuring equipment for determination of atmospheric content and spectra in orbital altitudes down to 120 kilometers. The system is to be capable of maintaining a fixed orientation and shall contain a propulsion system to allow orbital changes from parking orbits to low earth orbits. The system shall perform in space for a one year period.

3.1.1 OPERATIONAL

The Aeronomy Satellite shall be a self-contained payload system operating at nominal altitudes of between 120 kilometers and 800 kilometers. An onboard data system shall collect measurements at specified times and shall store such data until commanded to read out over a given ground station. A tracking transmitter shall be incorporated to provide orbital position information. The Goddard Space Flight Center Space Tracking and Data Acquisition Network (STADAN) shall be employed as the data receiving and command network. Solar cells and rechargeable batteries shall provide power to the spacecraft. The system shall interface structurally with the improved Thor Delta vehicle and shall interface electrically to the extent required for prelaunch electrical checkout and for physical separation from the vehicle subsequent to final stage burnout. Functional operations requirements are described in paragraph 3.3.

3.1.1.1 MISSION OPERATIONS

The Aeronomy Satellite shall be launched from an improved Thor Delta vehicle. The final stage shall place the payload in the parking orbit of 250 km perigee and 800 km apogee. At completion of this maneuver the payload will be separated from the last stage by means of a spring ejection mechanism. If spin stabilization is desired the spin rate will be provided by a spin table on the last stage prior to separation. Post launch checkout by ground command will be accomplished prior to mission maneuvers. The payload is required to maneuver into a low earth orbit with a 120 km perigee. A minimum of 10 operational passes through perigee will be performed during which all experiment sensors are operating below 240 km. Payload programming will be performed to allow storage of data for a period of 25 minutes during each operation with a subsequent playback over a pre-selected station in a time period of 2.5 minutes. Payload programming will be updated or changed from ground commands as necessary. Velocity increments, angular increments and time -to-thrust increments will be commanded from selected ground stations and shall be derived from ground analysis of orbital parameters and attitude information. At the conclusion of the required operational passes, the payload will be propelled to the parking orbit for full recovery of power systems and post-sequence checkout. A minimum of 24 maneuvers into low orbits shall be performed during the one year required lifetime. A minimum of 10 operational perigee passes will be performed with each maneuver. Positive termination of spacecraft transmission shall be provided, on an automatic timing basis. A command channel shall be set aside to provide a backup capability to interrupt power to both transmitters. The mission operations sequence is shown on Page I-27.

3.1.1.1.2 MISSION SUPPORT OPERATIONS

The STADAN system shall be utilized for tracking, data acquisition, and command. The following stations are required for support from a Cape Kennedy launch: Fort Meyers, Florida; Johannesburg, South Africa; Mojave, California; Woomera, Australia; Quito, Ecuador; Lima, Peru; Santiago, Chile; and Rosman, North Carolina. Data acquisition during launch operations will require support from the Air Force Eastern Test Range. Support requirements will be specified in a support requirements document. A Pacific coast launch to a near polar orbit will require tracking, data, and command support from all available STADAN stations. The Western Test Range shall be required for data acquisition support during launch operations.

The Aeronomy Satellite prime contractor shall be responsible to GSFC for Flight readiness tests, post installation tests and satellite launch preparations. GSFC shall be responsible for coordination of all pad activities concerning the satellite and shall commit the satellite for flight.

Orbital information must be analyzed to determine the required velocity, direction, and time durations for maneuvers to operational altitudes. Such parameters must be transmitted to the spacecraft via the STADAN command link at several stations. Additional analysis is required once the new orbit is established to set time increments into the payload programmer for power on, power off, record and reproduce functions. Timing must coincide with events such that experiment data is recorded during perigee passes and reproduction of data occurs over a suitable ground station. The mission support operations profile is depicted on Page I-28.

3.1.1.2 LOGISTICS

The Aeronomy Satellite must be transported from contractor's site to Goddard Space Flight Center, to a designated STADAN station for compatibility testing, and to the launch area. At the launch area, spare parts provisioning must be provided on-site for all minor assemblies, e.g., batteries, transmitters, etc, and for all expendables. Scheduling shall provide availability of major components as needed, e.g., motors, structure, etc. from fabrication and test cycles of next full assembly for shipment and replacement while undergoing launch operations. During operations at other than launch area, the spacecraft will not carry spares, but shall have at least minor component replacements available at the prime assembly area for shipment as required. The last flight will require spares provisioning on all parts at the replacement level with the exception of the basic structure. Operation at all sites requires availability of propellant. Environmental control will be provided at all times, including times of transit, and specifically will consist of air filtering, cooling, and moisture control.

3.1.1.3 PERSONNEL AND TRAINING

Ground operation of the Aeronomy Satellite will be accomplished by prime contractor personnel. Two major crews will be required: a four man crew will take charge of all transportation and movement of the vehicle; a four man crew will be responsible for all testing and maintenance of systems other than propulsion and A.C.S. Two additional persons are required for propellant loading and engine operations. One operations manager and an assistant will be assigned during launch and testing operations. All above personnel will be furnished by the prime contractor and will be in-plant trained for their specific operations responsibilities.

Goddard Space Flight Center will supply: computation personnel for orbital operations analysis; ground operations manager, networks controllers, communications engineers, STADAN personnel, program manager, launch director, and associated staff members.

3.1.2 PROGRAM DEFINITION

Development items are defined for this specification as those items which must be completely designed or which must undergo extensive redesign or tailoring to meet the objectives. Development items for the Aeronomy Satellite are:

1) Solar panel system
2) Structure and attachment
3) A.C.S.
4) Propellant storage and feed system
5) Power and command distributor
6) Signal conditioner
7) Ejection system
8) Wiring harnesses
9) Antenna

All other items are developed or may be tailored with minimum modification.

3.1.2.1 SYSTEM ENGINEERING DOCUMENTATION

The basic specifications tree is shown on Page I-29 and includes system, subsystem, and equipment specifications. The final study report under NAS8-18029 portrays the functional relationships of all systems and equipment.

3.1.2.2 PRIMARY FUNCTIONAL AREA LIST

Primary functional areas for the Aeronomy Satellite are:

1) Experiment
2) Data Handling
3) Communications
4) Power
5) Propulsion and A.C.S.
6) Ground Checkout

Requirements for these areas are specified in paragraph 3.3.

3.1.2.3 CONTRACT END ITEM LIST

CEI NO.	NOMENCLATURE	NEXT ASSEMBLY
SS-AE1000	Aeronomy Experiment System	Payload Support System
SS-AE2000	Payload Support System	Thor-Delta
SS-AE3000	Electrical Support System	N/A
SS-AE4000	Mechanical Support System	N/A

3.1.3 OPERABILITY

3.1.3.1 RELIABILITY

The airborne system reliability goal shall be .80 for an operating time of 1 year in space (8760 hours). The prototype system shall demonstrate 30 days accumulative running time during ground tests. The flight system shall demonstrate 14 days operation during ground tests.

3.1.3.2 MAINTAINABILITY

3.1.3.2.1 MAINTENANCE REQUIREMENTS

Construction of the Aeronomy Satellite will be modular, as concerns the electronics, such that each minor component (transmitter, encoder, etc.) maybe replaced without changing basic specifications nor altering data format, unless such change has associated with it an identifiable calibration tabulation that may be compensated for in-data reduction. Each component shall be completely identifiable and each change formally documented so that the exact serial number configuration and change ramifications are known at time of flight and, in fact, at every test point in the program. Battery packaging shall be especially designed for ease of access and removal, since this component is particularly time and useage limited. Add-on modularity shall be accommodated in the structural bolt patterns for the addition of experiment elements as required, and for the addition of power and signal cable harnessing to complete the additions. Subcommutation modules in the encoder system shall be provided on an optional basis for such changes that might warrant inclusion of additional data. These requirements dictate the use of measurement, power, and command distribution and the inclusion of spare wires in each harness from the distributors. A ten percent spare wire per major harness is a goal for design flexibility.

Accessibility shall be provided by design for replacement of components while in installed launch configuration. Such access shall be in effect until'area clear' is established prior to launch.

3.1.3.2.2 MAINTENANCE AND REPAIR CYCLE

As stated above, the minor component level shall establish the replaceability. Engine replacements shall require removal to the contractor facility; this shall require no longer than 1 week from start of removal until reestablishment to the same level of flight readiness ineffect at time of removal. Replacement of any minor component shall not delay launch nor the initiation of terminal countdown by more than 8 hours.

Repair of the removed component shall not exceed 24 hours unless an additional spare is made available or unless exchange is made during the day of launch. One battery pack shall be maintained in a state of readiness and shall replace the test battery as late as possible in the vehicle countdown preferably no earlier than 6 hours before scheduled launch. A second battery shall be available as standby in the event of evident of flight battery degradation. Battery replacement and continuity tests shall not require more than 2 hours.

3.1.3.3 USEFUL LIFE

The intended useful life of the satellite is 12 months (8760 hours). Positive termination of RF radiation shall be provided on an automatic basis. Ground command backup shall be provided for termination.

3.1.3.4 NATURAL ENVIRONMENT

3.1.3.4.1 LAUNCH

This section intentionally left blank, to be completed upon launch vehicle selection.

3.1.3.4.2 IN-FLIGHT ENVIRONMENT

Pressure: 10^{-3} to 10^{-7} Torr

Temperature: $-20^{\circ}C$ to $+70^{\circ}C$

Radiation:

Electrons 1 Mev or less @ 10^{9} electrons/cm^2/day
Protons 100 Mev or less @ 10^{6} protons/cm^2/day
UV 10^{-3} to 0.4 microns @ 10^{-13} to 10^{-5} ergs/cm^2/day
X-Ray 10^{-8} to 10^{-3} microns @ 10^{-5} ergs/cm^2/day

3.1.3.5 TRANSPORTABILITY

The weight of the Aeronomy Satellite shall not exceed 1200 pounds (544.3 kilograms). Installation to the vehicle will require a lifting crane suitable for above weight with safety margin. Movement between facilities shall be accomplished by truck or rail. Loading and unloading will require hydraulic lift decks with suitable securing devices. Packaging for shipment of the satellite shall be such as to provide portable environmental conditioning in transit. Suitable instrumentation shall be provided during travel to assure that thermal and structural stresses are within design specifications. Duration of exposure to uncontrolled environment shall not exceed one hour and protective covering shall be provided against precipitation during such exposure.

3.1.3.6 SAFETY

3.1.3.6.1 ORDNANCE DEVICES

All ordnance items to be used as part of the program shall be of a gas generating nature. Each squib shall have arming and safing plugs at the squib and be accessible for continuity checkout and firing tests. The safing plug shall place an electrical short circuit across the bridge wires and shall open circuit the power lead. The arming plug shall provide proper connection for firing, but the firing circuitry shall be such as to maintain the shorted bridge and open firing line until proper command for firing is issued. There shall be two bridge wires associated with each squib and each ordnance function will contain two squibs. Dual firing circuits and dual isolated ordnance batteries shall be incorporated such that operation of either battery, either firing circuit, and either bridgewire of either squib will accomplish the required action. Redundant barometric switches will be incorporated into each battery's firing line for protection against premature firing until a specified altitude is reached.

3.1.3.6.2 PROPELLANT

A monopropellant system shall be used for primary propulsion during orbit changes. Pressurized explusion of gas shall be used for attitude correction and precession of spin axis. Premature ignition of the propellant shall be prevented by design of the igniter system such that the catalyst shall be maintained in a closed container with solenoid valve power open circuited during all operations until commanded to fire. Propellant loading will require payload area clearing of non-essential personnel and will be accomplished as the last payload operation prior to igniter setting and area closeout.

3.1.3.6.3 SAFETY WIRING

All threaded parts of the Aeronomy Satellite which are likely to work loose in service shall be adequately secured. Whenever practicible, all threaded parts shall be safety wired in accordance with MS33540.

3.1.3.7 INDUCED ENVIRONMENT

The Aeronomy Satellite shall be designed to dissipate 10 watts electrical power maximum, during launch operations until shroud ejection. Fifteen watts average electrical power shall be dissipated when power from the solar array is available. Thermal design shall provide temperature control to the battery of from $0^{\circ}C$ to $40^{\circ}C$ Celsius while operating from the natural in-flight environment of paragraph 3.1.3.4.2. Solar array design shall be such as to provide an upper temperature limit at the solar cells of $+40^{\circ}C$ while operating in direct sunlight for 54.3 minutes, nominally, and in cold space for 39.3 minutes, nominally, per orbital period. All equipment other than those described above shall be capable of operation from $-10^{\circ}C$ to $+70^{\circ}C$ and thermal design shall be such as to provide that amount of control when operating in the natural environments of paragraph 3.1.3.4.2, in solar cycles as described above, and in engine induced temperatures.

The engine installation and surround shall be such that an average of 180 BTU/sec shall be rejected from the satellite during the engine operation periods, without exceeding the temperatures specified for the above locations.

3.2 SYSTEM DESIGN AND CONSTRUCTION STANDARDS

3.2.1 GENERAL DESIGN AND CONSTRUCTION STANDARDS

3.2.1.1 SELECTION OF SPECIFICATIONS AND STANDARDS

All Standards or Specifications other than those established and approved for use by NASA must be approved by the procurring agency prior to incorporation into any system, component, or facility specification.

3.2.1.2 MATERIALS, PARTS, AND PROCESSES

Specifications and Standards for all materials, parts, and government certified an approved processes and equipment not specifically designated in this specification and which are necessary for execution of this specification shall be approved by the procurring agency prior to use and shall be included as revisions to this specification. Workmanship specifications as pertains to soldering, welding, encapsulation, conformal coating, etc. shall not be designated in this specification but shall be designated in all lower level specifications; e.g., subsystem/equipment specifications.

3.2.1.3 STANDARD AND COMMERCIAL PARTS

Standard parts (AN, MS) shall be used wherever suitable for the purpose and shall be identified on the drawings by standard part number. Commercial utility parts such as screws, nuts, bolts, cotter pins, etc. may be used providing they possess suitable properties and are replaceable by standard parts (AN, MS) without alteration; and further, providing the corresponding standard part numbers are referenced in the standard parts list, and if practicible, on the vendor's drawings.

In the event there is no suitable corresponding standard part in effect, commercial utility parts may be used, providing they conform to all requirements of applicable specifications.

3.2.1.4 MOISTURE AND FUNGUS RESISTANCE

Materials that are nutrients to fungi shall not be used when it is practical to avoid them. If used, the material shall be treated with a fungicidal agent capable of meeting the environmental requirements of this specification. The use of materials which are nutrients for fungus shall not be prohibited in hermetically sealed assemblies and other accepted and qualified uses. Non-soluable parts shall be used throughout where practicible; in the event that the use of soluable parts is practically unavoidable, such parts shall be properly treated or sealed to prevent decomposition.

3.2.1.5 CORROSION OF METAL PARTS

All metals used in construction of equipment, mechanical interfaces, and associated connections covered by the equipment specifications of this system shall be protected to resist corrosion during normal service life. Dissimilar metals, as defined by MS 33586, shall not be used in intimate contact unless suitably protected against electrolytic corrosion per MSFC-SPEC-250.

3.2.1.6 INTERCHANGEABILITY AND REPLACEABILITY

All parts having the same manufacturer's part number shall be directly and completely interchangeable with each other with respect to installation and performance. Change of manufacturer's part number shall be governed by the drawing number requirements of specification MIL-D-70327.

3.2.1.7 WORKMANSHIP

The equipment affected by this specification, including all parts and accessories, shall be constructed, assembled, and finished in a thoroughly workmanlike manner. Particular attention shall be given to neatness and thoroughness of painting, welding, brazing, riveting, wire-stripping, soldering, machine screw assemblage, and marking of parts and assemblies. Burrs, paint chips, or loose parts shall be considered workmanship defects. Workmanship standards and specifications shall be cited in all lower level specifications.

3.2.1.8 ELECTROMAGNETIC INTERFERENCE

The design of the integrated Aeronomy Satellite shall be such that the completed unit meets the requirements of MIL-E-6051C when coupled with the launch vehicle. Each system of the satellite shall meet the requirements of MIL-I-6181D.

3.2.1.9 IDENTIFICATION AND MARKING

Each component of the satellite shall be marked in a permanent and legible manner in accordance with MIL-STD-130. The permanency of the marking shall at least equal the life expectancy of the equipment. Decalcomanias and stencils are not acceptable.

3.2.1.10 STORAGE

The Aeronomy Satellite shall have a storage life capability of no less than 8 months without part replacement or maintenance, other than propellant and attitude control gasses, which must be installed prior to launch. Batter storage of 2 months shall not be exceeded without reconditioning.

3.2.2 DESIGN ENGINEERING AREAS

All airborne and ground support electrical and electronic equipment shall comply with the minimum requirements of MSFC-PROC-256.

3.3 REQUIREMENTS FOR FUNCTIONAL AREAS

3.3.1 EXPERIMENT

The specific functional requirements for the aeronomy experiment mission will be specified by the GSFC. General requirements are that the experiment will occupy less than 3460 cubic inches (56, 699 cubic centimeters), and will require 10 watts of power on a per day average basis assuming operation of 2 hours per 24 hour period.

3.3.2 DATA HANDLING

The airborne data handling system shall operate at a bit rate of 8, 640 bits per second and shall contain a minimum sampling rate for a given experiment of 180 sps. Thirty analog housekeeping measurements of 1/3 sps minimum shall be accommodated. The system shall consist of:

a) Measurement Distributor
b) Signal Conditioner
c) PCM Data Encoder
d) Data Switch & Programmer Logic
e) Magnetic Tape Recorder

Operation of the system shall be such as to provide real time sequencing of data through the data switch to the low power communications link. The same output may be, upon command, applied to the magnetic tape recorder for storage at the 8, 640 bits per second rate for a period not exceeding 25 minutes. Readout of stored data will take place over a STADAN station at a rate of 86, 400 bits per second in a period not exceeding 2.5 minutes. Programming of specific times for record and playback will be accomplished through the Data Switch and Record Programmer Logic by way of ground command through the Communications System.

Signal conditioning will be centralized to provide 0 to 5 VDC analog levels and 0 ± 1 to 5 ± 1.5 VDC digital levels for multiplexing in the data encoder. A Measurement Distributor will centralize all transducer outputs for application to the Signal Conditioner. Signal commons shall be above chassis ground and isolated from the power source common. Single point grounding shall be accomplished in the Communications System at the transmitter. Signal conditioning shall provide a measurement accuracy at least equal to the full scale analog resolution of the encoder (8 bits, 5 volts, .39%).

The Data Handling system shall consume a power of 1.52 watts computed on an "ON" time average, with a peak power of 9.2 watts. Daily average power consumption shall be 1.52 watts.

3.3.3 COMMUNICATIONS

The airborne transmission system shall operate at assigned frequencies in the 136 to 137 mHz band. Two frequencies shall be assigned; one shall be assigned for a tracking transmitter, and one shall be assigned for an FM data transmitter. The tracking (PM) transmitter shall be capable of modulation by the 8640 bits per second output from the digital data encoder (when so commanded) while maintaining sufficient power at the carrier for tracking within a maximum slant range of 540 N. miles (1000 kilometers) from the STADAN interferometer system. Total power, radiated, from the PM transmitter shall be sufficient to provide acceptable ground receiver signal to a 30 KHz bandwidth from a slant range of 1520 N. miles (2810 kilometers) corresponding to a horizon elevation of five degrees at the 430 N. mile (800 kilometer) maximum apogee. The duty cycle for the PM transmitter operation shall be assumed to be 100 percent at full power.

Total radiated power from the FM transmitter shall be sufficient to provide acceptable ground receiver signal to a 300 kHz predetection bandwidth from a slant range of 1520 N. miles (2810 kilometers) corresponding to a ground station horizon elevation of five degrees at the maximum apogee condition. The FM transmitter shall be commanded for turn-on and for modulation and shall operate for three minutes while being modulated by the 86,400 bits per second output of the magnetic tape recorder. Average useage shall be assumed to be two hours per twenty-four hour period.

Minimum FM transmitter radiated power shall be 2 watts. Minimum PM transmitter radiated power shall be 0.1 watts.

A command receiver/decoder system compatible with the STADAN 148 mHz, 70 channel command system shall be incorporated. An antenna system with suitable mixers and separators shall be employed for transmission and reception to provide isotropic pattern coverage.

The communications functional requirements shall be met with the following equipment:

a) FM Transmitter
b) PM Transmitter
c) Command Receiver
d) Command Decoder
e) Diplexer and Filter
f) Hybrid Ring
g) Antennas

3.3.4 POWER

Electrical power shall be provided by solar cells and rechargeable Nickel-Cadmium batteries. Equipment load power requirements shall be assumed to be 15 watts maximum including all converter and regulator losses. Sufficient battery capacity shall be available to allow powering of this load for a continuous period of 25 minutes without recharging and while discharging to a maximum depth of 10 percent of rated capacity. N/P solar cells of eleven percent efficiency at air mass "1" shall be mounted on the exterior surface of the payload structure in an optimum serial/parallel arrangement. Projected area of the cells and corresponding power output shall be sufficient to replace 140 percent of the energy taken from the batteries during a nominal effective sunlight period of 54.3 minutes per orbit. Solar power availability calculations shall include derating of solar cells from a 140 mw/cm^2 solar input for air mass zero operation (85% efficiency), operational solar cell temperature (50^{o}C), radiation losses over one year's operation (5%), mismatch in electrical connection (8%), glass transmission loss (8%), and air mass "1" cell efficiency. Per preliminary design reported in section 6, of the final report the cells shall be mounted around the payload structure to provide a minimum projected solar cell area to the sun of 2,800 square centimeters. A value of 8.68 milliwatts per square centimeter shall be used for power calculations at 40 degrees Celsius. Daily power average shall be assumed to less than 15 watts with a peak power of 150 watts for 2 hours. Minimum battery capacity shall be 20 ampere hours. The operational and functional requirements of the power system shall be met with the following equipment:

a) Solar Array
b) Battery Charger
c) Battery
d) Converter
e) Regulator
f) Power and Command Distributor

3.3.5 PROPULSION AND REACTION CONTROL SYSTEM

3.3.5.1 The propulsion system shall be capable of multiple restarts till exhaustion of the propellant supply. The target mission requires 75 operational restarts: the unit will be demonstrated for acceptance by imposing 125 restarts with 30 minute cooling periods between operational restarts.

Minimum duration per restart shall not be less than (0.267 Isp) seconds for the propellant system used, referred to altitude. The system shall be capable of consuming 750 lbm propellant mass during the specified sequency of restarts, for the target mission engine. Other design consumptions will be specified by pursuing detailed tradeoff studies indicated in this report.

The valve system reliability shall be 0.999 or better for the mission; parallel redundant systems shall not be required unless demonstrably necessary.

The tankage shall consist of appropriately passivated materials to minimize propellant decomposition during orbit life. Pressurization levels shall be nominally 250 psia, and proof pressures shall be 750 psia. Burst pressure shall be 1000 psi. The filled and pressurized tanks and distribution manifolding shall be capable of operating between $-10^{\circ}C$ and $+70^{\circ}C$. Tank pressurization shall be by a properly regulated and filtered gaseous nitrogen supply; propellant shall be settled by centrifugal force without bladders for spin stabilized satellite design; and by positive bladder expulsion for other designs.

3.3.5.2 REACTION CONTROL SYSTEM

The reaction control system for the spin stabilized satellite shall be sized to precess the spin axis 90° in less than 5 minutes. The system shall consist of a single thruster operating from the main engine propellants. The thruster and associated propellant control valve shall be capable of delivering 6000 operational restarts in a pulsing mode with 0.9999 reliability; the unit will have a demonstrated capability of completing 20,000 pulse cycles without failure.

The pulse duration shall not be more than 0.1 second, during while not less than 2 lb-seconds of effective impulse shall be delivered. The thruster system shall provide a total impulse of 750 lb-seconds.

The reaction control system for the local vertical stabilized satellite design shall utilize gaseous nitrogen from a regulated pressurization source separate from the main propulsion system pressurization.

The reaction control system for the Aeronomy/550 design is a mass expulsion type using gaseous nitrogen as the working fluid. A diagram of the RCS is shown on Page I-30 and the positions of the nozzles on the vehicle are indicated on Page I-31. The system uses eight thrusters, two in-pitch, two in-yaw, and four in-roll. The same limit cycle period and maneuvering characteristics shall obtain on all axes.

The gas supply is contained in two 900 cubic inch tanks at the aft end of the vehicle. This position permits minimum piping requirements. When charged with nitrogen at 3000 psi (20.7×10^2 N/M^2) and 70° F the tanks contain approximately 9 lbm (4.08 kg) of gas. Leakage shall not exceed 60 std. cc/hr.

The thrusters are supplied with 15 psia nitrogen from a pressure regulator and are activated by solenoid valves upon command from the attitude control system. Off-the-shelf parts developed for the Mariner may be used.

3.3.6 GROUND CHECKOUT

Functional requirements for ground checkout are to provide electrical checkout of the satellite during test operations, to provide electrical checkout of the satellite during blockhouse operations, and to provide mechanical handling capabilities for transportation and vehicle installation. Electrical test checkout shall provide fault isolation down to the minor component (replacement) level and shall include capabilities for RF reception, data processing, data display, and data recording. Blockhouse checkout shall provide fault isolation down to the system functional level (e.g. data, communications, power, experiment, propulsion, and A.C.S.) Both blockhouse and test checkout functions shall include open loop, closed loop, and hardline RF by-pass capabilities. Mechanical handling requirements shall include environmental control of encased payload, portability of encased payload, and provision of handling attachments (lift rings, etc). Interface requirements include facilities interfaces with hydraulic decks and handling cranes for vehicle installation.

4. QUALITY ASSURANCE

4.1 PHASE I, INTEGRATED PROJECT TEST REQUIREMENTS

4.1.1 ENGINEERING TEST AND EVALUATION

Functional testing shall be conducted on the integrated Aeronomy Satellite to provide verification of the functional requirements of paragraph 3.3. Electrical checkout test equipment shall be utilized to verify interface and functional compatibility, and it shall have been acceptance tested and calibrated prior to use on the satellite. In general, the functional tests shall verify the following:

a) Command response to simulated STADAN commands
b) Power dissipation and voltage generation
c) Frequency and radiated RF power

d) Bit rate and bit rate stability
e) Magnetic tape loading and playback
f) Safety of ordnance and propellant
g) Automatic response to attitude change while in cruise mode (angular change check only)
h) Attitude change angle response to ground command (angular change check only)
i) Propellant ignition actuation (W/O propellant)

These tests shall be conducted on both the prototype and flight units at room controlled temperature, pressure, and humidity. Except for tests of b. above, all tests will be performed using GSE supplied power at maximum and minimum expected voltage variations. In the case of the prototype satellite, all applicable functional requirements shall be verified before proceeding to formal qualification. The flight satellite shall complete the above engineering test and evaluation prior to formal acceptance testing.

4.1.2 QUALIFICATION TESTING

Formal qualification tests will be performed on the integrated prototype satellite and experiment. These tests shall verify the functions of paragraph 4.1.1 and shall be confined to the following environments:

a) Thermal-vacuum
b) Vibration
c) Voltage variations
d) Radio frequency interference
e) Thermal shock
f) Acceleration
g) Humidity
h) Acoustic Noise

The payload shall be required to function within specification during all tests with the exceptions of Thermal Shock and Humidity, and shall be required to function within specification before and after exposure to all tests. Selected portions of MIL-STD-810A shall be used for Thermal Shock, Humidity, Thermal-Vacuum, and Acceleration. MIL-I-6171D shall be used as the reference for the Radio Frequency Interference Test.

4.1.2.1 QUALIFICATION TEST CONDITIONS

Qualification test conditions are to be further ammended to include launch vehicle test environmental specifications at the time of vehicle selection.*

(1) Thermal-vacuum: The Aeronomy Satellite and experiment shall be subjected to temperatures ranging from -10^{o} to $+70^{o}$ Celsius while operating in simulated altitudes of sea level to 200,000 feet (60,960 meters).

(2) Thermal Shock: The system shall be subjected to temperatures changing from $+85^{o}$ to -40^{o} Celsius in a period of not less than 2.5 nor more than 4 minutes, and from -40^{o} to $+85^{o}$C in the same time interval.

(3) Humidity: The system shall be subjected to a relative humidity of 95% for a period of 5 hours with the temperature gradually increasing from $+25^{o}$ to $+70^{o}$ Celsius.

(4) Acceleration: The system shall be subjected to accelerations of * times the acceleration due to gravity for 3 minutes in both directions along the three major axes.

(5) Sinusoidal Sweep Test: The system shall be subjected to frequencies from 5 Hz to 2 kHz and from 2 kHz to 5 Hz at the rate of one octave per minute, at test levels as furnished by GSFC.

(6) Random Vibration Test: The integrated airborne system shall be subjected to random vibrations for five minutes at test levels furnished by GSFC.

(7) Acoustic Noise: The integrated airborne system shall be subjected to acoustic noise levels up to 143 db in a frequency

range of 5 Hz to 10 kHz with a reference level of 2×10^{-5} N/M^2. Test level limit values will be furnished by GSFC.

4.1.3 ACCEPTANCE TESTING

Formal acceptance testing shall be conducted on the integrated flight satellite and experiment. The acceptance test shall provide verification of the functions listed in paragraph 4.1.1 and shall be conducted under combined conditions of: a) Temperature (-10° to $+70^{\circ}$ Celsius); b) Altitude (sea level to 200,000 feet 60,960 meters); and c) Voltage variations.

4.1.4 LAUNCH VEHICLE COMPATIBILITY TEST

The integrated prototype Aeronomy Experiments Satellite representing flight configuration, shall be installed on an improved Thor Delta launch vehicle and shall undergo testing to verify compatibility. This test shall verify mechanical and electrical interfaces and shall establish RF compatibility along the general lines of MIL-E-6051C. The degree of compliance (or demonstration thereof) shall be established by GSFC.

4.1.5 STADAN COMPATIBILITY TEST

The prototype Aeronomy Experiments Satellite, representing Flight Configuration, shall be transported by van to a designated STADAN station for basic compatibility testing. Tests shall verify tracking and data reception compatibility and shall include the transmission of specified commands to the satellite from the STADAN command system. Selected portions of the PM data will be displayed to simulate quick look operations.

4.2 PHASE II, INTEGRATED PROGRAM TEST REQUIREMENTS

4.2.1 FLIGHT READINESS TESTS

Flight readiness tests shall be performed with the Aeronomy Experiments Satellite installed on the launch vehicle.

Tests shall include RF compatibility; countdown demonstration (including battery installation, propellant loading, and area closeouts); simulated countdown and launch (with test batteries, no propellant, and including ordnance test chamber firing, payload spinup, and simulated ejection); and interface continuity tests (including no-fire squib continuity and no-fire catalyst valve continuity). Tests will be performed using blockhouse checkout consoles and suitcase continuity testers, where necessary.

4.2.2 COUNTDOWN TESTS

During terminal countdown, all systems with the exception of the propulsion and A.C.S. system, will be tested to assure performance prior to satellite launch commit.

4.2.3 RANGE TESTS

Prior to the initiation of final countdown, all participating data acquisition and tracking stations shall simulate reception of satellite data and tracking information and shall provide normal communications with GSFC/Greenbelt for satellite orientation and orbital control exercises.

5. PREPARATION FOR DELIVERY

Each deliverable contract end item shall be packaged at the contractor facility for delivery to a government specified facility. The container used for packaging of the Aeronomy Satellite shall contain a portable environmental conditioning system and shall provide handling devices for crane attachment. Construction of packaging for all CEI's shall be such that disassembly for inspection purposes shall not alter its protective capability upon reassembly. The shipping containers shall be marked in accordance with MIL-STD-129C and the following information:

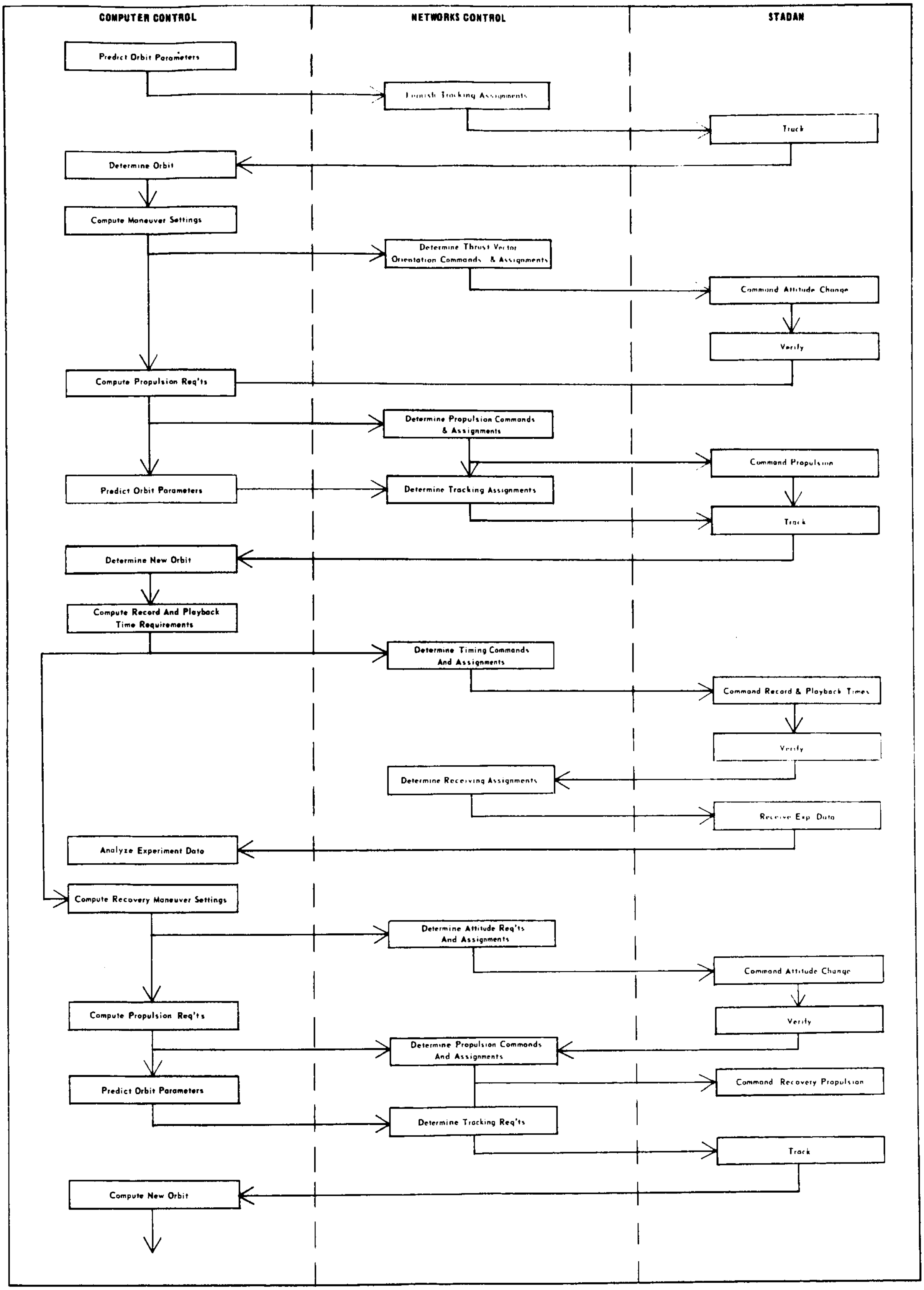
a) Nomenclature
b) Contract Number

c) Contracting Agency
d) Contractor Name
e) Contractor Trade Mark
f) Date Packaged
g) Inspection Date
h) Specification Number

6. NOTES

6.1 DEVIATIONS AND REVISIONS

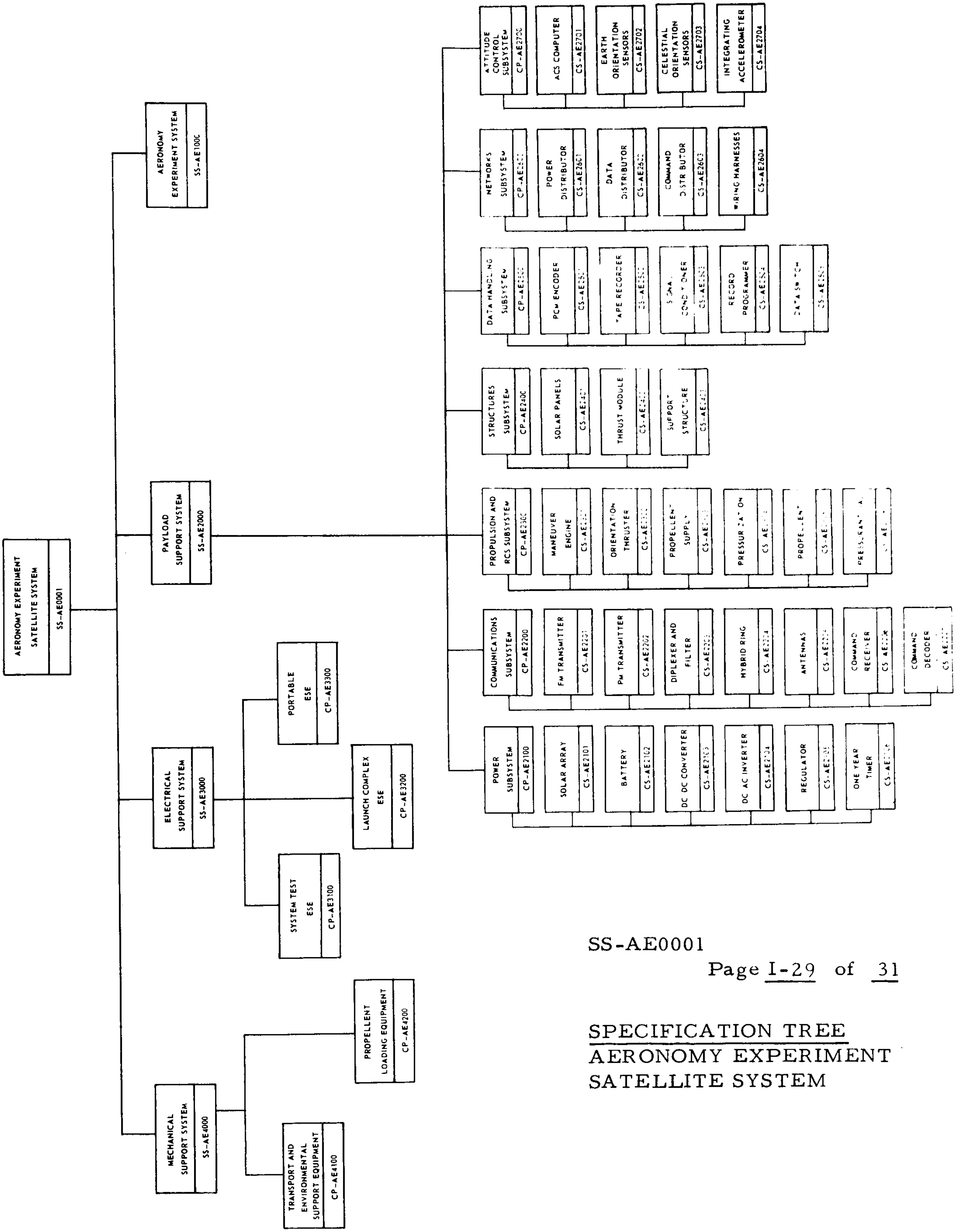
Deviations to this specification shall be published in an approved deviation appendix. Revisions shall be made only after formal approval of GSFC and will be submitted to all holders of specifications, along with a revision control sheet.



MISSION FLIGHT OPERATIONS SUPPORT SEQUENCE

MISSION OPERATIONS PROFILE

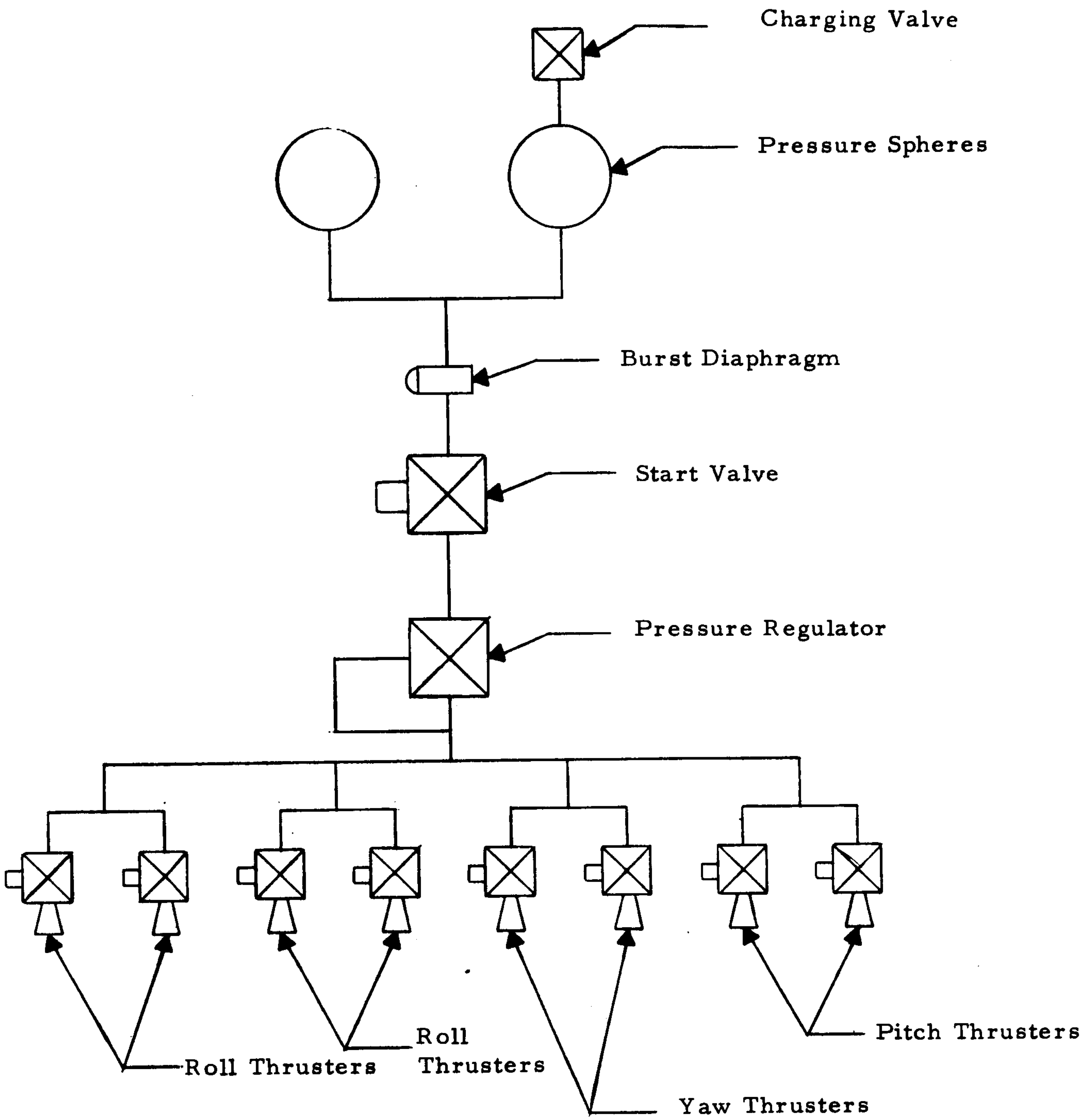
LAUNCH VEHICLE	SATELLITE					GROUND STATION			
	Propulsion & A.C.S.	PM TRANSMISSION	FM TRANSMISSION	COMMAND RECEPTION	DATA	TRACK	PM RECPT.	FM RECEPT.	COMMAND
		Transmit Real Time				X	X		
Provide Parking Orbit									
Establish Initial Attitude									
Eject Shroud									
Eject Satellite									
	Maintain Attitude								
				Receive Propulsion Settings					X
	Orient Thrust Axis								
				Receive Propulsion Execute					X
	Lower Perigee								
	Reestablish Attitude								
				Receive Timing Command					X
			Transmit Stored Data		Record Data			X	
			⋮		Reproduce Data			⋮	
				Receive Orientation Settings	⋮				X
				Receive Propulsion Sequence					X
	Orient Thrust Axis								
	Raise Perigee								
				Receive Orientation Settings					X
				Receive Propulsion Sequence					X
	Raise Apogee								
	Reestablish Attitude								



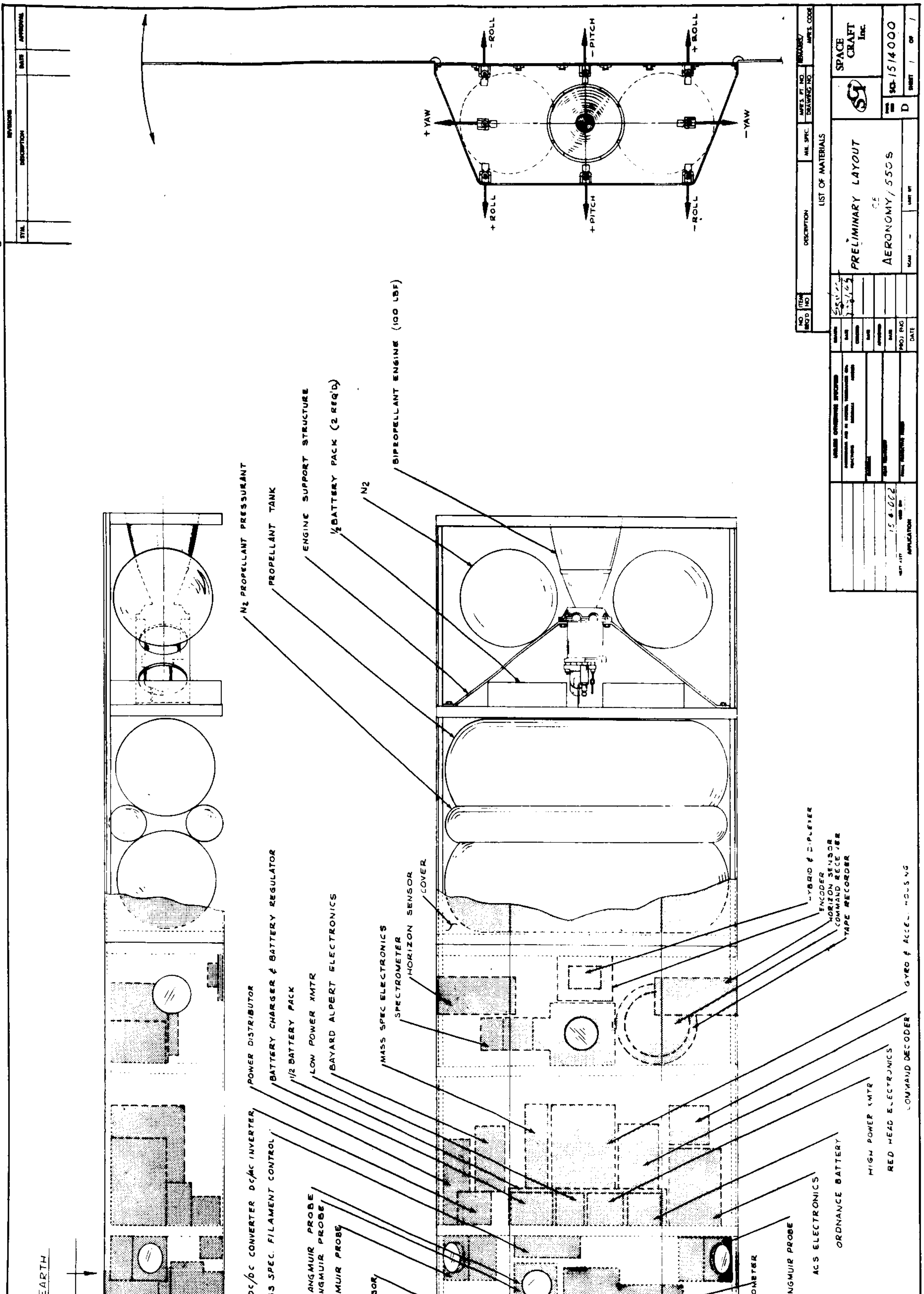
SS-AE0001

Page I-29 of 31

SPECIFICATION TREE
AERONOMY EXPERIMENT SATELLITE SYSTEM



SCHEMATIC, REACTION CONTROL SYSTEM


FLIGHT
EARTH
N_2 PROPELLANT PRESSURANT
PROPELLANT TANK
ENGINE SUPPORT STRUCTURE
1/2 BATTERY PACK (2 REQ'D)
N_2
BIPROPELLANT ENGINE (100 LBF)
+ROLL
-ROLL
+PITCH
-PITCH
+YAW
-YAW
COVER
HORIZON SENSOR
SPECTROMETER
MASS SPEC ELECTRONICS
BAYARD ALPERT ELECTRONICS
LOW POWER XMTR
1/2 BATTERY PACK
BATTERY CHARGER & BATTERY REGULATOR
POWER DISTRIBUTOR
DC/DC CONVERTER DC/AC INVERTER
MASS SPEC. FILAMENT CONTROL
(TOP) LANGMUIR PROBE
(BOTTOM) LANGMUIR PROBE
LANGMUIR PROBE
BAYARD ALPERT SENSOR
LANGMUIR PROBE
HORIZON SENSOR
RED HEAD SENSOR
SPECTROMETER
LANGMUIR PROBE
ACS ELECTRONICS
ORDNANCE BATTERY
HIGH POWER XMTR
RED HEAD ELECTRONICS
COMMAND DECODER
GYRO & ACCEL. HOUSING
TAPE RECORDER
COMMAND RECEIVER
HORIZON SENSOR
ENCODER
HYBRID & DIPLEXER
LIST OF MATERIALS
PRELIMINARY LAYOUT
AERONOMY/SSOS
SPACE CRAFT Inc.
D
SCI-1514000

PROGRAM PLAN AND COST ESTIMATES

7.1 PROGRAM PLAN

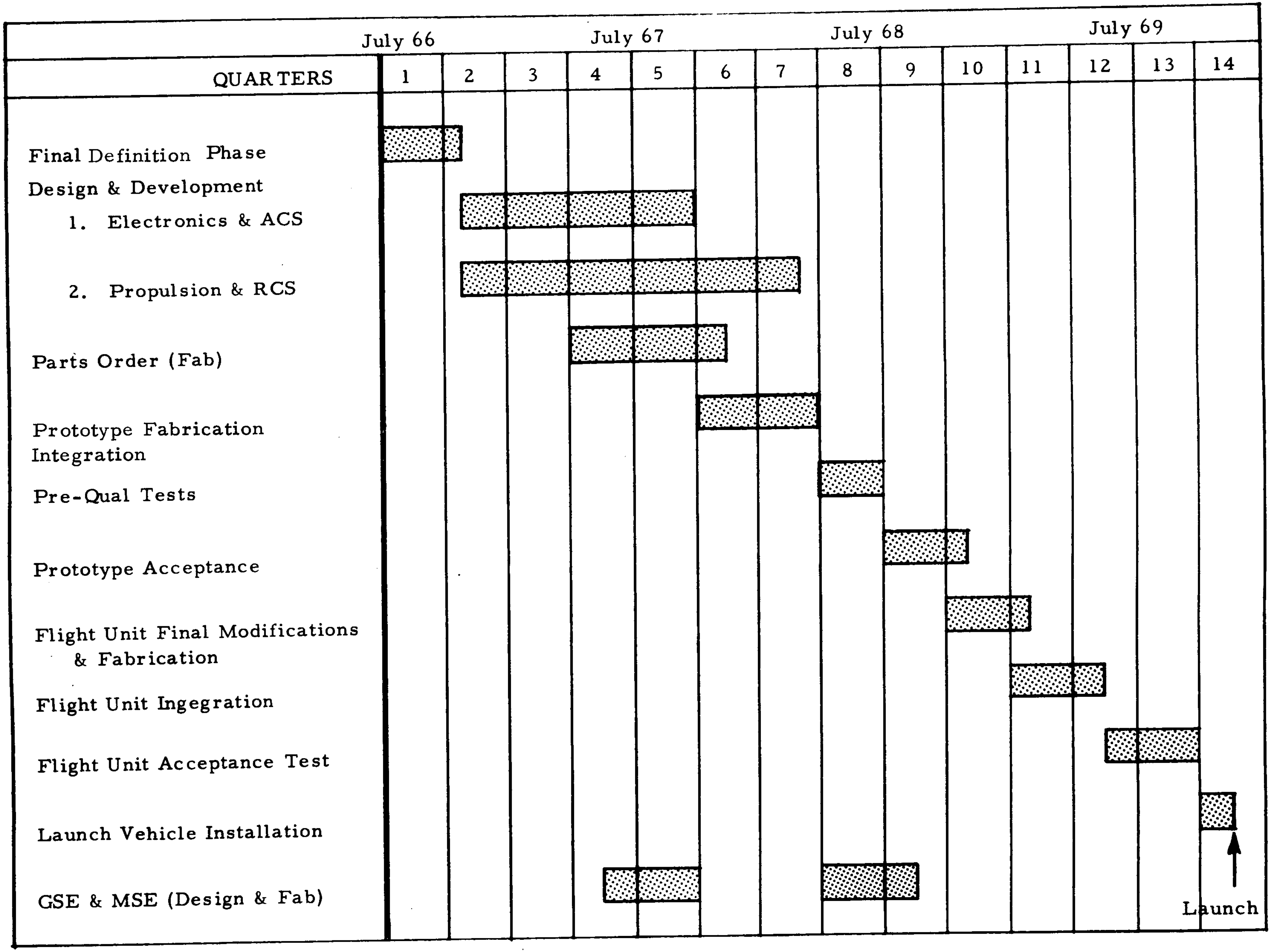
7.1.1 Final Definition Phase

The program plan scheduled in Figure 7-1 begins with a final definition study which selects the single approach from the two competing concepts found in the present report. Section 8, Recommendations, presents a more detailed work statement for this phase. Specifically, the system functional requirements, and system specifications are frozen to the level of detail necessary to write comprehensive subsystems specifications and to complete preliminary detail designs of the subsystems.

At the same time, cost elements are confirmed to the same level of detail. These will include costing (and scheduling) software such as preliminary and final specifications, test plans and documentation, and progress reports as well as the hardware items. Systems Engineering shall provide these configuration control documents in format sufficient for a procurement package during the final definition phase. In particular, the final definition should provide for purchase request definition for development of critical elements of the propulsion and reaction control system, and possibly the power system.

7.1.2 Design and Development

The engineering team completing the final definition study builds on that study after development contract award. Systems engineering begins in the final definition phase with an intense effort to validate all the system functional requirements and operational requirements that were developed in the preliminary studies. These requirements form a sound basis for the detail



MAJOR MILESTONE CHART

design and the detail definition of the interfaces between the system elements. Two system engineering activities are initiated immediately upon contract award. These are the final detailing and validation of the development plan and the detailing of facility and tooling requirements. The validated development plan provides hard schedule targets for all the subsystem developments, subsystem tests, and establishes cost for the accomplishment for each significant point of the program. This development plan will have established a specific subsystem schedule and component development schedule assigning a specific responsibility for every system element down to the component level.

Priority is given to the propulsion and ACS systems at first, since these have the longest lead times and are time-critical. The purchase descriptions for development contracts provided under Section 7.1 above are validated and procurement action initiated by issuing RFQ's. Selection and subcontract invitation should proceed during the second month of the contract period.

Since the final definition studies have given adequate information to define subsystems down to major component levels, all other major component procurements should occur during the fourth quarter. The key members of the team who will conduct the detail design have been involved in the detailing and validating of the system requirements and finalizations of the system and subsystem specifications. After specifications near completion, these individuals are shifted to the integrated detail design. The development design details are plowed back into finalization of the specification by these key individuals. This approach to the development insures rapid execution of the detail design since the nucleus of the design teams have intimate knowledge of all the system's functional requirements and are familiar with all system interfaces.

As overall design progresses, the propulsion and RCS elements are being tested at the subcontractor's plant. During this time, the electronic elements of the ACS breadboard will be operated to validate detail design. During later development tests the ACS is integrated with the command system and the reaction control system (RCS) response will be analoged and mounted in a mockup of the airframe for testing. Simulations of various flight phases will be conducted. These tests will simulate all the operational interfaces of the attitude control system. Inspection of the schedule shows that support equipment will be available in the latter portion of development, so that the final operational configurations tested will correspond to flight unit specifications.

The development phase is formally complete when partially packed subsystems have been tested as units against the non-complete Part II of specifications. In addition, breadboard harness hookups are used to test for preliminary integrated system response.

Careful attention to detail during these development tests will assure that the prototype fabrication and development tests can proceed with a minimum of changes to the subsystem. As a result of development testing, qualification testing can begin late in the eighth quarter.

7.1.3 Prototype Fabrication, Integration, and Pre-Qualification Tests

The development effort carried on from the second through the seventh quarters is paralleled with RFQ's and purchase orders beginning in the fourth quarter. Note that development item purchases are not broken out separately - the "parts order" referes to prototype hardware. Elements of the development breadboards are actually integrated into the prototype unit; the degree of such use will be determined on an individual component basis. Prototype assemblies will conform to final configuration control documents, and will reflect all changes made through the end of development phase testing. During the prototype phase, emphasis is placed on integrated systems response, and on preliminary environmental response tests for critical items. Electronics final package configurations are subjected to temperature and vacuum tests, and critical assemblies may be vibrated to launch vehicle conditions. Propulsion and RCS is packaged into the structure and preliminary mechanical integrity and leakage tests are conducted. When the prototype assembly is complete except for solar cell application, it is given a final check using support equipment, and passes to pre-qualification tests.

A solar panel assembly is separately tested for solar response and for mechanical, electrical and temperature integrity. A pre-prototype structure is outfitted for mechanical and electrical integration tests with the launch vehicle and handling gear.

Upon successful completion of the pre-qualification test phase, the equipment passes to the (probably) less severe environment of the prototype acceptance test.

7.1.4 Prototype Acceptance and Release for Flight Unit Fabrication

Prototype acceptance will be conducted to strict specification limits. Testing may take place in phases at separate locations to be established; e.g., contractor's plant for functional simulation phases, propulsion subcontractor for corresponding tests, and GSFC for space simulation phases.

7.1.5 Flight Unit Fabrication, Integration and Acceptance Tests

Modification indicated by the prototype acceptance tests are incorporated into the configuration control documentation prior to fabrication. Fabrication utilizes the same tools and jigs used for the prototype. The quality control functions called out in the specifications will be carefully implemented to assure that only flight grade material is accepted and incorporated into the flight unit.

Flight unit integration will begin shortly after manufacturing operations are initiated. These tests will validate all the subsystem interfaces and the operation of the system under ambient conditions. A series of systems tests completes the pre-qualification of the flight unit. The first flight article will be completed at the beginning of the eleventh quarter and integrated during the early twelfth quarter. The completion of integrated systems tests permits flight unit acceptance tests to be initiated. Four to five months are allowed for this phase to permit scheduling of facilities and to allow for random scheduling difficulties. Tests follow the prototype pattern.

7.1.6 Flight Unit Satellite Installation and Checkout

The flight systems checkout at the launch area requires from four to six weeks. Integration will be supported by the integration contractor, the propulsion subcontractor, and a GSFC team. The developed and delivered support equipment will be fully utilized in the launch vehicle integration and checkout phase.

7.1.7 Ground Support Equipment

Preliminary details design of the ground support equipment and mechanical support equipment will be set forth during the final definition studies which preceed this development. Initiation of the detail design of ground support equipment is therefore deferred until the detail design of the spacecraft and all its subsystems are essentially complete. The ground support equipment design will then be completed by selected members of the engineering team which executed the detailed spacecraft design. Design and fabrication of the two basic groups of ground support equipment will be accomplished during the final development phase of the electronics and ACS. Ground support equipment is then available for subsystem and the integrated system tests of the prototype.

7.2 COST ANALYSIS FOR AN AERONOMY/SS SATELLITE

The cost analysis presented in this section is intended to serve as a coarse guideline only. It represents a rapid estimate of probable prototoype satellite and support equipment costs based upon a number of more detailed estimates made over the past two years for satellites containing similar subsystems. The numbers should be used with caution until further effort is made to substantiate them. Table 7-1 indicates an estimated cost of $2,685,000 for a development program delivering one prototype satellite and one set of support equipment. A continuation of the program to deliver a flight satellite and another set of support equipment is estimated to cost $1,800,000. The delivered hardware cost is therefore $4,485,000, exclusive of government management, engineering and operations costs.

A brief definition of the cost categories is given to aid in interpreting the estimates. Hardware costs provide for delivery of off-shelf items and for tailoring to specifications of components required for development items. The subsystem development costs are carried in the column, "Development". The integration of subsystems into the major systems, as well as integration into the satellite, are listed as 'Satellite Integration'. These three major categories of electronics, propulsion, and attitude control are tested as entities under "Satellite Tests". Systems engineering costs are here defined to include essentially all software management and engineering costs: contractor program management, cost control, schedule control, configuration control including documentation and specifications, test management and documentation, and progress reports.

Prototype costs can, in our opinion, be reduced for the first flight item by considering that hardware costs should fall slightly, development cost can be reduced by perhaps 75 percent, (the remaining 25 percent being used to accommodate design changes), integration costs fall only slightly, test costs

DEVELOPMENT THROUGH PROTOTYPE ONLY

Subsystems	Hardware	Development & Test			System Engineering	Totals
		Development	Satellite Integration	Satellite Tests		
Data	80.0	95.0				
Communications	16.0	20.0	200	180	175.0	
Power	206.0	20.0				
Propulsion	85.0	350.0	200	100	95.0	
ACS RCS	80.0	150.0	100	70	95.0	
Cabling	4.0	9.0		3.0	2.0	
	471.0	644.0	500.0	353.0	367.0	2, 335
ESE	100					
GSE						
Propulsion & RCS	75					
ACS Similation & Checkout	150					
MSE						
Dollies, Tools, Hoists, Tables	25					
	350					350
Subtotal, Prototype Development (Thousands)						2, 685
DEVELOPMENT OF FLIGHT UNIT						
Flight Unit	450	100	450	250	200	
Support Equipment	350					
Subtotal	800	100	450	250	200	1, 800
PROGRAM TOTAL (THOUSANDS)						$4, 485

are less because procedures are firm, and systems engineering costs could fall to about 50 percent of the prototype values. The support equipment figures remain identical on the basis that a second set of equipment is desirable since three locations will share its use: GSFC, the integrating contractor, and KSFC. These considerations are reflected in the flight unit costs of Table 7-1.

RECOMMENDATIONS

Preliminary estimates were made of the cost effectiveness of the several approaches. The basis selected for the comparison was daily cost based on at least one year of operation. Payloads met the basic mission definition during the year, except for the Scout payload. Under these rules, the two competing 1200 lbm concepts show greatest cost effectiveness. It is recommended that, unless strong experiment configuration or operational factors are discovered, the further studies be limited to the Aeronomy/SS 1200 lbm concepts and the Aeronomy/550T concept.

The salient differences between the recommended designs are the essentially untried two-pulse ACS system for the Aeronomy/SS; and the esentially untried periodically stepped solar panels for the Aeronomy/550T. A selection between these concepts should properly rest on the criteria of cost, reliability, and mission profile flexibility. It is recommended that combined satellite and operations sequence reliability estimates be assessed for both designs. Attention should be focussed on the overall ACS-RCS performance and reliability, including the net operations and on the solar panel actuator performance where used. In addition, a survey should be made under this task of the pros and cons of each configuration in mission profiles not studied to date. The objective of this part of the definition is to identify mission profiles for which a given configuration is best suited: it has already been established that either configuration can accomplish the defined basic mission.

The resulting tradeoff between system reliability and optimum mission profiles should be definite enough to allow selection of a simple design approach.

The results of the preceding two recommendations will be a single design approach. The third recommendation encompasses a more detailed study of the mission profiles for the selected configuration only. Section 4 of this report presents ample evidence of the need for more detailed parametric analysis, in particular tradeoffs between apogee, parking perigee and working perigee; effects of inclination changes and lift vehicle margins; and varying schedules within an established orbit using an updated atmosphere.

As the most advantageous mission profile emerges, the comparisons will also seek to establish reasons for selecting either a monopropellant or bipropellant system. This selection can again be aided by combining cost effectiveness and reliability estimates. Detailed characteristics of the selected configuration are then based on the findings of this recommended task.

Following completion of the third recommendation, detail preliminary design can be initiated. This considers first, confirmation of all constraints not specifically detailed in the preceding tasks; and secondly, confirmation of the detail design approach indicated in the present study. Following this phase, it is recommended that attention be focused on improving the mass fraction of the satellite to an extent consistent with cost and present technology estimate, confirming same with vendors, and arriving at specification limits. This effort places first emphasis on propellant tanks, second emphasis on structure. Subsystem masses should be confirmed but are considered to be of less importance.

The final recommendation is to perform Systems Engineering tasks to the depth necessary to assure GSFC that project mileposts and costs are realistic; this entails vendor/subcontractor cost confirmation, and identification of critical items, and a detailing of the development costs indicated in section 7.

Early execution of the recommended tasks completes the major portion of the final definition phase indicated in the program plan; figure 7-1, and will permit early initiation of the actual satellite development phase.

APPENDIX A

RELATIONSHIP BETWEEN DESIGN PARAMETERS AND MISSION PROFILE

The total lift-off weight off the spacecraft may be determined by

$$W_t = W_{str} + W_{pp} + W_{ta} + W_{ex} + W_{ss} + W_{acs} \qquad (1)$$

where W_t = total lift-off weight

W_{str} = structure weight

W_{pp} = propellant weight for propulsion

W_{ta} = propellant tank weight

W_{ex} = experiment weight

W_{ss} = support system weight

and W_{acs} = weight of ACS expendables if a mass expulsion system is used.

It has been found for the basic structure being considered the structure weight is directly proportional to the total weight and the two may be related by

$$W_{str} = K_{str} \; W_t \qquad (2)$$

The tank weight for propellant stored under pressure is a function of tank volume and may be obtained by

$$W_{ta} = D \; \frac{(W_{pp} + W_{acs})}{\rho} \qquad (3)$$

where ρ in the propellant density in lbm/ft^3 and the weights are in lbm. D is a **realistic** tank weight per unit volume .

Assuming a mass expulsion Reaction Control System as described in Section 5.2.7 the weight of propellant required for attitude control may be determined as follows.

The turning increment per pulse is given by

$$\Delta\Theta = \frac{F\,d\,\Delta t}{I\,\omega} \tag{4}$$

where

$\Delta\Theta$ = turning increment,

F = RCS engine thrust,

d = moment arm of the thruster,

Δt = pulse duration,

I = moment of inertia of the spacecraft about its spin axis,

and ω = spin rate.

Assuming a homogeneous mass distribution

$$I = \frac{1}{2}\,\frac{W\,r^2}{g_c} \tag{5}$$

where

r is the spacecraft radius.

If d is assumed equal to r combining (5) and (4) results in

$$F\,\Delta t = \frac{\Delta\Theta\,W\,r\,\omega}{2\,g_c} \tag{6}$$

The total $\Delta\Theta$ for each maneuver is 3π radians, therefore, the total impulse required for positioning during each maneuver is

$$(F\,\Delta t)_n = \frac{3\pi\,W\,r\,\omega}{2\,g_c} \tag{7}$$

where W is the average weight of the spacecraft during the maneuver. The ACS propellant consumed during N maneuver is then given by

$$W_{acs} = \frac{3 \pi r \omega N}{2g_c (I_{sp})_{acs}} \left[\frac{2 W_t - W_{pp} - W_{acs}}{2} \right] \tag{8a}$$

where the quanity in the brackets is the average weight of the spacecraft after N maneuvers and $(I_{sp})_{acs}$ is the effective specific impulse of the RCS propellant.

Rearranging terms in (8a) it is found that

$$W_{acs} = \frac{3 \pi r \omega N}{2} \left[\frac{2 W_t - W_{pp}}{2g_c (I_{sp})_{acs} + 3 \pi r \omega N} \right] \tag{8}$$

Substituting (2), (3), and (8) into (1) the expression

$$W_t = \frac{\left\{ \left[1. + \frac{D}{\rho} - \frac{3 \pi r \omega N \left(1. + \frac{D}{\rho} \right)}{2 (2g_c (I_{sp})_{acs} + 3 \pi r \omega N)} \right] W_{pp} + (W_{ex} + W_{ss}) \right\}}{\left\{ 1. - K_{str} \frac{3 \pi r \omega N \left(1 + \frac{D}{\rho} \right)}{(2g_c (I_{sp})_{acs} + 3 \pi r \omega N)} \right\}} \tag{9}$$

is obtained.

The propellant weight required for N maneuver may be determined approximately by

$$W_{pp} = W_t \left[1 - exp \left(\frac{-N \Delta V_m}{I_{sp} g} \right) \right] \tag{10}$$

It should be noted that equation (10) is in fact in approximation since it was derived from the rocket equation, i.e.

$$\Delta V = I_{sp}\, g_c \ln \frac{W_t}{W_t - W_{pp}} \tag{11}$$

and in the derivation of (11) it is assumed that the only mass lost is that of propellant used in producing ΔV and therefore equation (10) will yeild somewhat conservative values for W_p. The use of equation (10) is justified by the resulting simplicity (?) and the fact that atmospheric drag is being neglected in this analysis. This assumption offsets the error introduced by the equation (10) to some extent.

Combining (9) and (10) and solving for W_t

$$W_t = \frac{(W_{ex} + W_{ss})}{\left\{ -K_{str}\frac{D}{\rho} - \frac{3\pi r \omega N\left(1. + \frac{D}{\rho}\right)}{2\,(2g_c\,(I_{sp})_{acs} + 3\pi r \omega N)} + \left[1. + \frac{D}{\rho} - \frac{3\pi r \omega N\left(1. + \frac{D}{\rho}\right)}{2\,(2g_c\,(I_{sp})_{acs} + 3\pi r \omega N)}\right] \exp\left(\frac{-N\Delta V}{I_{sp}\, g_c}\right)\right\}} \tag{12}$$

Equation 12 relates design and mission parameters. (Remember that drag has been neglected).

APPENDIX B

SATELLITE GEOMETRY

B.1 EQUATIONS AND OPTIMIZATION

The general equation for the reduced volume of symmetrical tank configurations exactly enclosed in a right circular cylinder is given. The quantities are shown in Figure B-1. N primary tanks are mutually tangent near the cylinder's center. These are represented by the first term on the right of the equation. N secondary tanks are represented by the second term; these exist only if $\gamma \geqslant \mathrm{CSC}\,\pi/N$, and next between and tangent to the primary tanks. All tanks have equal radius.

$$KV = N\left(\frac{4}{3}\mu + \gamma\right) R^{-\frac{3}{2}} + N\left(\frac{4}{3}\mu + \gamma - \mathrm{CSC}\,\alpha\right) R^{-\frac{3}{2}}$$

Where

$$K = \frac{1}{\pi}\left(\frac{A}{4}\right)^{-\frac{3}{2}}$$, A being the satellite cross sectional area

N = number of primary tanks

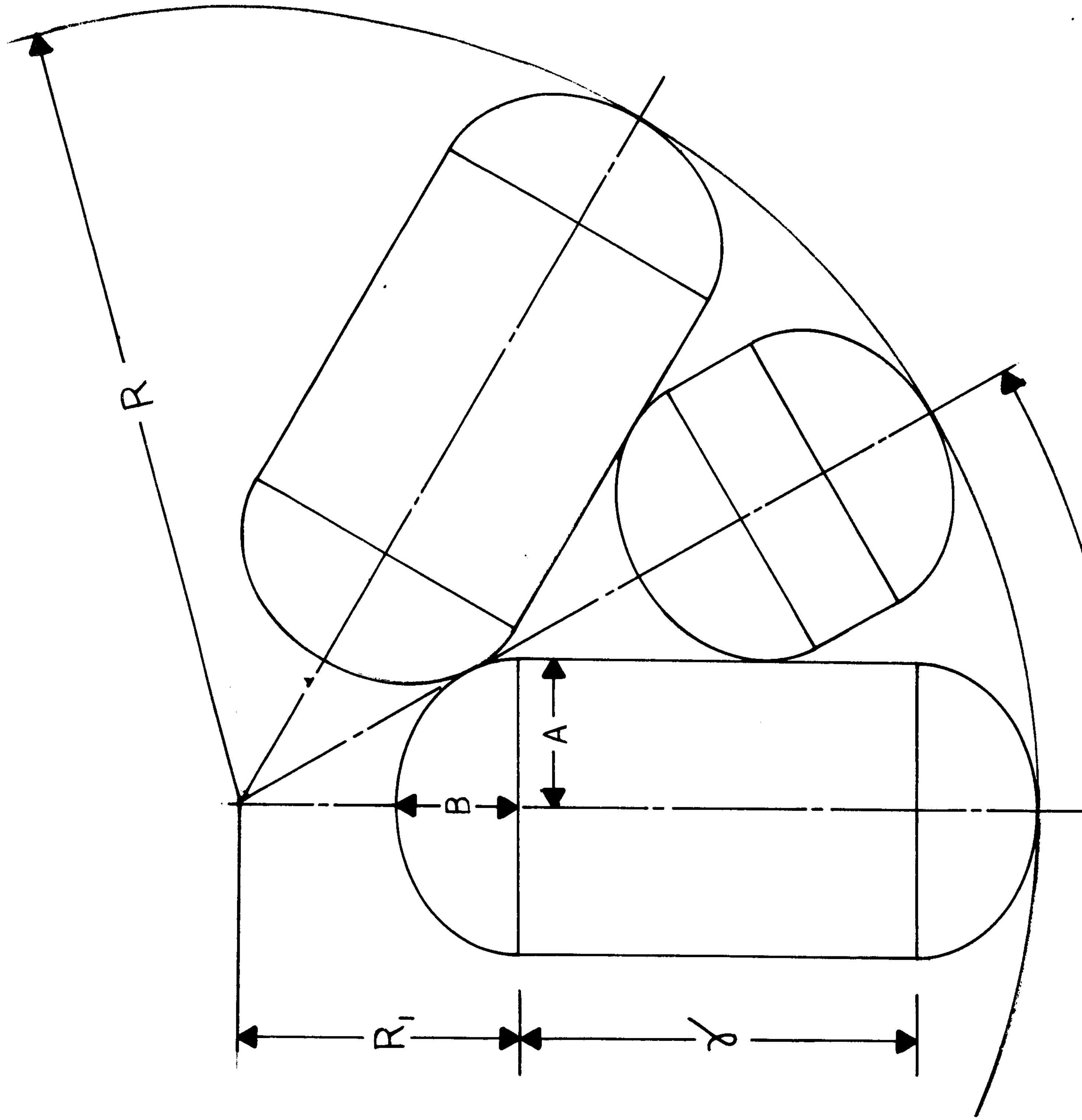
$R = R_1 + \mu + \gamma$ = satellite reduced radius

$\gamma = \ell/a$, the ratio of primary tank cylindrical length (between ellipsoid cartesian centers) to radius

$\alpha = \pi/N$, half the satellite central angle subtended by a primary tank.

$R_1 = \left(\mathrm{ctn}^2\alpha + \mu^2\right)^{\frac{1}{2}}$, the reduced radius from the satellite center to the ellipsoid cartesian centers

The tables of Section B-2 are computed using this equation for geometric volume.



DEFINITION OF TERMS

FIGURE B-1

Several 'optimum' criteria are examined. These are based on the knowledge that the highest practicable initial ballistic coefficient, $\beta = W/C_D A$ gives longest satellite life. This applies to inert as well as mass expulsion systems and holds for all propellant/total mass functions based on equivalent structures technology. Since propellant represents about half of the initial satellite mass for cases in this report, investigation of optimum storage criteria and the ratio V/A is of some interest. The ratio increases its effect on mission life as the mission time in the higher drag regions is extended.

B. 1. 1 Does an N exist which optimizes the V/A ratio for spheres? Neglecting the secondary tank term:

Eq B2 $$K\left(\frac{\partial V}{\partial N}\right)_A \equiv 0 = R - \frac{3\pi}{2} N^{-1} R_1^{-1} \operatorname{ctn}\delta \csc^2\delta$$

which has a solution for $\mu = 1$, $\gamma = 0$ of

Eq B3 $$N = \frac{3\pi}{2} \frac{\operatorname{ctn}\delta}{1+\sin\delta}$$

evaluation:

7 = 6.84 , e = -0.16

8 = 8.23 , e = +0.23

and the optimum number of spheres to minimize V/A lies between 7 and 8.

B. 1. 2 Since optimum V/A does not correspond to an integral number of tanks, does a γ exist that provides optimum V/A for a given N?

Solving $K\left(\frac{\partial V}{\partial N}\right) \equiv 0$ for γ, leaving $\mu = 1$,

Eq B4 $$\gamma = 3\pi N^{-1} \operatorname{ctn}\delta \csc\delta - \csc\delta - 1$$

The solutions for the optimum elongation for spherical end tanks are shown in Figure B-2.

Note that no solution exists for negative values of gamma. The first integral solution of $\gamma = 0.1$ approximately appears at N = 8.

B.1.3 Examination of the basic equations factor $\left(\frac{4}{3}\mu + \gamma\right)$ would indicate that for a fixed satellite radius, it is profitable to increase μ at the expense of γ.

However, for a fixed R

$$R = R_1 + \mu + \gamma = (\mathrm{ctn}^2 \alpha + \mu^2)^{\frac{1}{2}} + \mu + \gamma = \mathrm{CONST}$$

$$\frac{\partial \gamma}{\partial \mu} = -\left(\mu R_1^{-1} + 1\right)$$

Using

$$K \left(\frac{\partial V}{\partial N}\right)_{N,R} \equiv 0 = K \frac{\partial V}{\partial \mu} + K \frac{\partial V}{\partial \gamma} \frac{\partial \gamma}{\partial \mu}$$

and from the basic equation

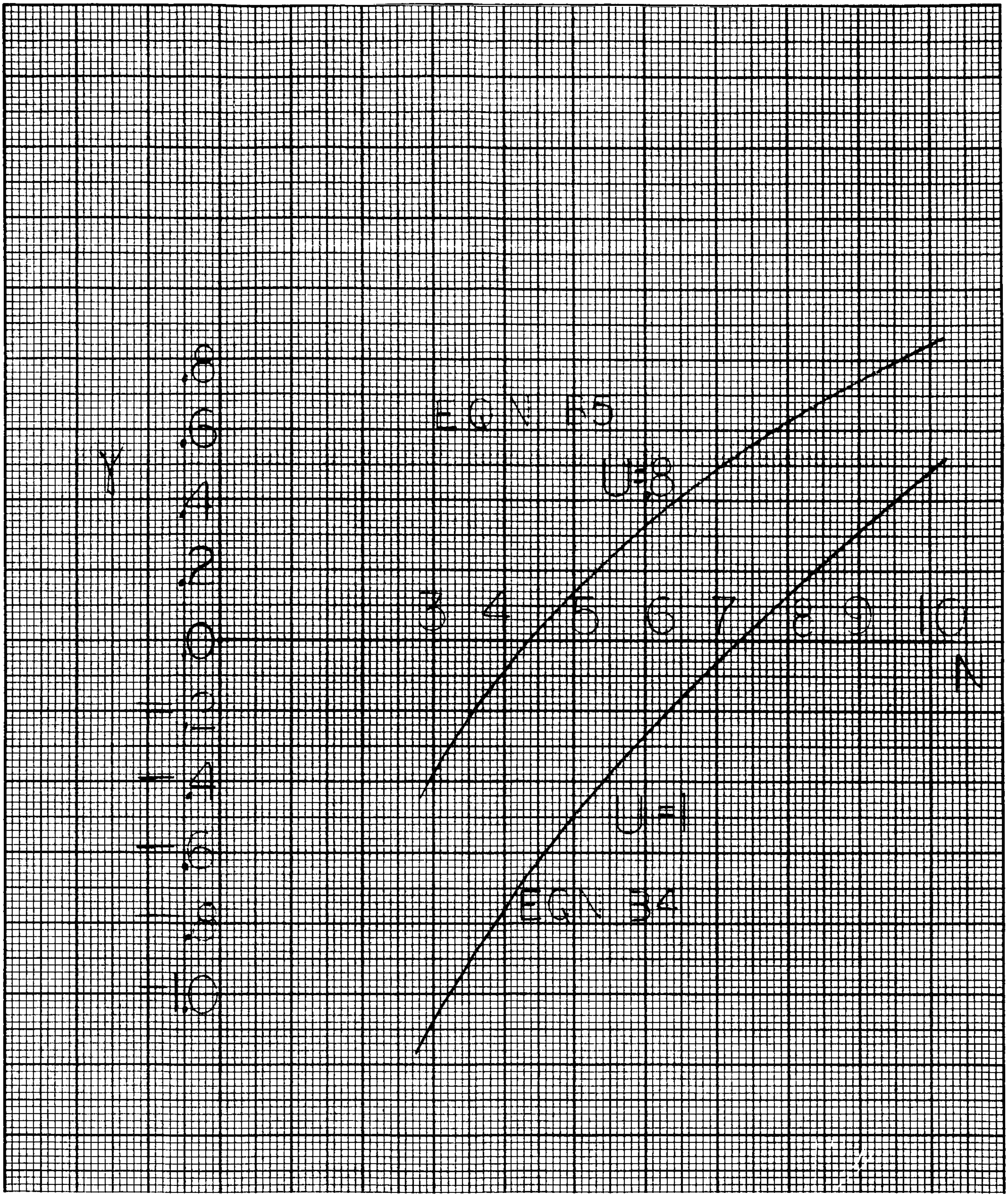
$$K \frac{\partial V}{\partial \mu} = \frac{4}{3} R^{-3/2}$$

$$K \frac{\partial V}{\partial \gamma} = R^{-3/2}$$

Substituting the partial differential values and simplifying, one obtains

$$\mu = \frac{1}{3} R_1 = \frac{1}{3}\left(\mathrm{ctn}^2 \frac{\pi}{N} + \mu^2\right)$$

or $\mu = \sqrt{8}\ \mathrm{ctn} \frac{\pi}{N} = 2.83\ \mathrm{ctn} \frac{\pi}{N}$



γ FOR N TANKS TO YIELD OPTIMUM V/A

FIGURE B-2

That is, the tank head should intersect the radius to the ellipse center at the 1/3 point from the latter: this is impossible since μ has a maximum value of 1 (spherical head) so that $\mu \leqslant .353$, or

$$120^\circ < \text{ctn}^{-1} .353 < 180^\circ$$

and $N < 3$, a value of no practical interest. There is no optimum for μ, but minimizing it to a practical extent improves V/A for elongated tanks. A value of $\mu = 0.8$ is chosen for the following calculations and tables.

B.1.4 What is the effect of $\mu = 0.8$ on the optimum $\gamma(N)$? Equation B3 is valid if we again neglect the secondary tanks: substitution for R and R_1 and rearrangement yields:

Eq B5 $$\gamma = \frac{3\pi}{2} \frac{\text{ctn}\,\delta}{N \sin^2 \delta} R_1^{-1} - R - \mu$$

or $$\gamma = \frac{3\pi}{2} N^{-1} R^{-1} \text{ctn}\,\delta \csc^2 \delta \left(\text{ctn}^2 \delta + \mu^2\right)^{-\frac{1}{2}} - \left(\text{ctn}^2 \delta + \mu^2\right) -$$

The function is plotted in Figure B-2, and shows that an optimum γ exists for $N > 4$.

This concludes analysis of tank systems with N tanks.

B.1.5 Does an optimum $\gamma(N)$ exist for the tank nests described by equation B1? Differentiating the full equation B1, and equating to zero results in

Eq B6 $$0 = \left(\frac{8}{3}\mu - \csc \delta + 2\gamma\right)\left(1 - \frac{3\pi}{2} \frac{\text{ctn}\,\delta \csc \delta}{N R_1 \sin \delta} R^{-1}\right) - \frac{\text{ctn}\,\delta \csc \delta}{N}$$

The explicit solution for γ is

Eq B7 $$\gamma = \frac{-b \pm \sqrt{b^2 - 4ac}}{2a}$$

where

$$a = 2B$$
$$b = CB - A - 2 + 2B\,(R_1 + \mu)$$
$$c = (CB - A)\,(R_1 + \mu) - C$$
$$A = R_1 \sin\alpha$$
$$B = NR_1 \sin^2\alpha \quad \tan\alpha$$
$$C = \frac{8}{3}\mu - \csc\alpha$$

The function is plotted in Figure B3, and shows that solutions exist for N greater than 5.

B.1.6 Since $\partial V/\partial\mu \doteq 0$ has no solution and $\partial V/\partial N$ does not lead to an optimum gamma within the design range, does an optimum tank radius to satellite radius ratio exist for an arbitrary choice of N and μ ?

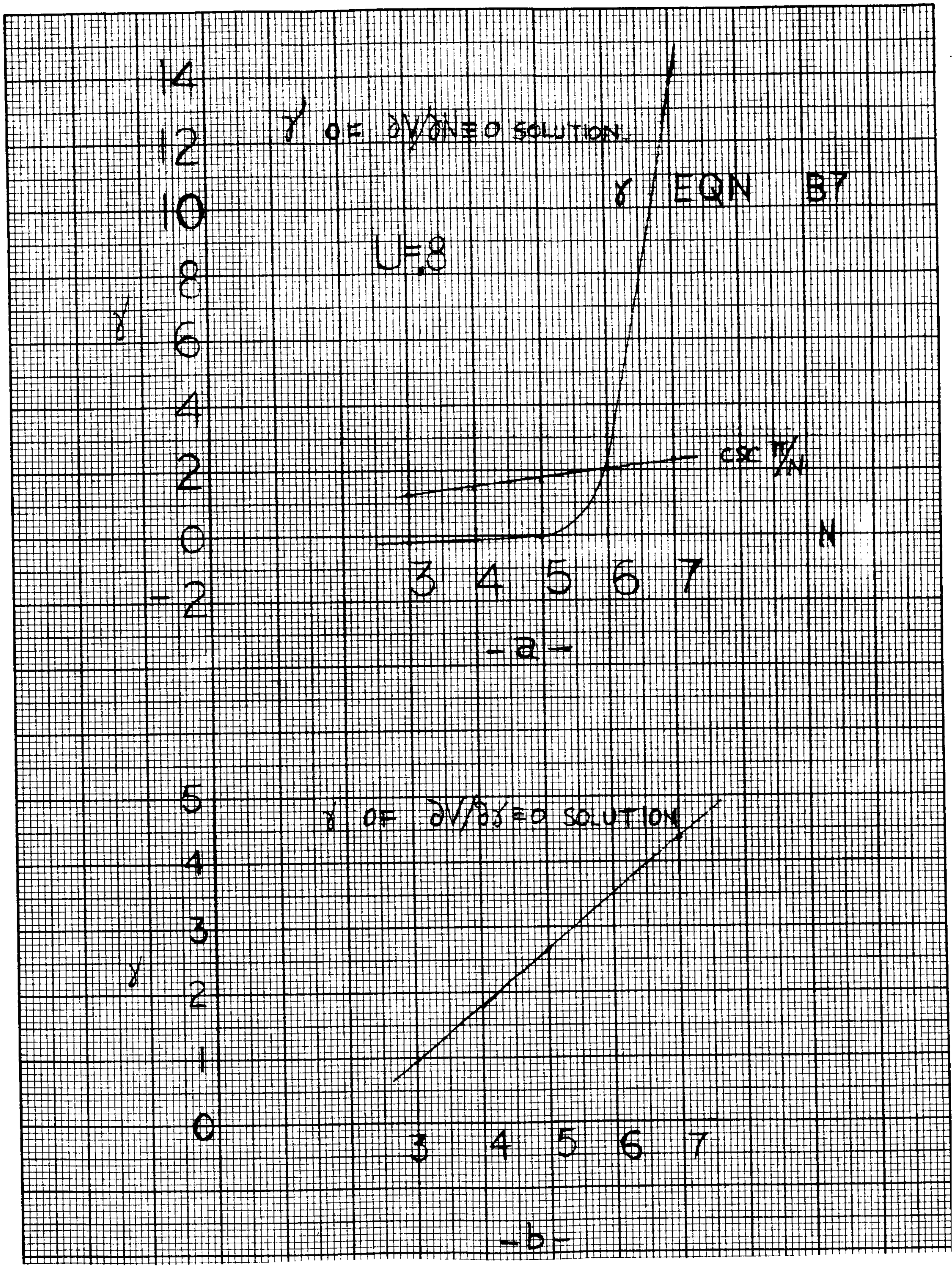
The solution to $K\,\partial V/\partial\gamma$ is

Eq B8 $$\gamma = R_1 + 3/2 \csc\alpha - 2$$

and is plotted in Figure B3b for N + N nests. Optima exist. The tables B1 and B2 use an adjusted volume which shifts the γ values upward about one unit.

B.1.7 Summary of Optima

Equations B3, B4, B5, and B7 either have no extreme values or these are out of the design range; nor can an optimum μ be determined. A methodical



γ FOR N + N TANKS, NESTED, TO YIELD OPTIMUM V/A
FIGURE B-3

search for a best configuration is the final solution, except that for each N, equation B8 indicates the optimum gamma for that N, in the sense of minimizing frontal area for unit propellant volume.

B.2 TABULATED VALUES

Equations 6 & 7 of Section 5.2.6 relate the geometric and the net volume of propellant tanks to the propellant and tank mass. Equations of this appendix relate geometric volume to satellite minimum dimensions. Table B1 and Table B2 record characteristics of nested tank designs for hydrazine propellant. The following nomenclature applies:

N	number of long tanks
A	satellite minimum cross sectional area
G	gamma, distance between tank head epicenters
SATR	satellite radius
TKR	tank radius
TKW	tank mass
WP	propellant mass
W	sum of last two
WPTOA	figure of merit, = WP/A
C_1	R_1 of Figure B1
C_2	corresponding radius to secondary tank
PHI	Propellant mass in a high volume tank
PLO	Propellant mass in a low volume tank
RP	Ratio of propellant masses
BA	ratio of head minor to major diameter
ROM	tank material density
FT	virtual tank density: lbm/ft^3
PL & PH	propellant density
Unit Basis:	ft. and lbm.

BA = 0.80 ROM = 180.00 FT = 16.10 L = 1 PL = 62.40 PH = 62.40

N	A	G	SATR	TKR	TKW	WP	W	WPTOA	C1	C2	PHI	PLO	RP
3	3.50	1.50	1.696	0.516	80.	274.	354.	78.	0.51	1.10	61.	30.	0.5
4	3.50	1.50	1.770	0.494	87.	297.	384.	85.	0.63	1.33	54.	21.	0.4
5	3.50	1.50	1.845	0.474	69.	237.	306.	68.	0.75	1.56	47.	0.	0.0
6	3.50	1.50	1.919	0.456	74.	253.	327.	72.	0.87	1.78	42.	0.	0.0
3	4.00	1.50	1.813	0.552	98.	335.	433.	84.	0.54	1.18	75.	37.	0.5
4	4.00	1.50	1.892	0.528	106.	363.	469.	91.	0.68	1.42	66.	25.	0.4
5	4.00	1.50	1.973	0.507	85.	289.	374.	72.	0.81	1.67	58.	0.	0.0
6	4.00	1.50	2.051	0.487	90.	309.	399.	77.	0.93	1.91	51.	0.	0.0
3	5.00	1.50	2.027	0.617	137.	468.	605.	94.	0.61	1.32	104.	52.	0.5
4	5.00	1.50	2.116	0.591	148.	508.	656.	102.	0.76	1.59	92.	35.	0.4
5	5.00	1.50	2.206	0.567	118.	404.	522.	81.	0.90	1.87	81.	0.	0.0
6	5.00	1.50	2.293	0.545	126.	432.	558.	86.	1.04	2.13	72.	0.	0.0
3	5.50	1.50	2.126	0.647	158.	540.	698.	98.	0.64	1.39	120.	60.	0.5
4	5.50	1.50	2.219	0.620	171.	586.	757.	107.	0.79	1.67	106.	41.	0.4
5	5.50	1.50	2.313	0.594	136.	466.	603.	85.	0.95	1.96	93.	0.	0.0
6	5.50	1.50	2.405	0.572	145.	498.	643.	91.	1.09	2.23	83.	0.	0.0
3	6.00	1.50	2.220	0.676	180.	616.	795.	103.	0.67	1.45	137.	68.	0.5
4	6.00	1.50	2.318	0.647	195.	667.	862.	111.	0.83	1.74	120.	46.	0.4
5	6.00	1.50	2.416	0.621	155.	532.	687.	89.	0.99	2.04	106.	0.	0.0
3	6.50	1.50	2.311	0.703	203.	694.	897.	107.	0.69	1.51	154.	77.	0.5
4	6.50	1.50	2.412	0.674	220.	753.	972.	116.	0.86	1.82	136.	52.	0.4
3	3.50	2.01	1.823	0.480	82.	281.	363.	80.	0.47	1.03	59.	35.	0.6
4	3.50	2.01	1.892	0.462	92.	315.	408.	90.	0.59	1.25	53.	26.	0.5
5	3.50	2.01	1.963	0.446	96.	329.	425.	94.	0.71	1.47	47.	19.	0.4
6	3.50	2.01	2.032	0.431	96.	329.	425.	94.	0.82	1.68	43.	12.	0.3
3	4.00	2.01	1.948	0.513	100.	343.	443.	86.	0.51	1.10	72.	42.	0.6
4	4.00	2.01	2.023	0.494	113.	385.	498.	96.	0.63	1.33	64.	32.	0.5
5	4.00	2.01	2.098	0.477	117.	402.	520.	101.	0.76	1.57	58.	23.	0.4
6	4.00	2.01	2.172	0.460	117.	402.	519.	100.	0.88	1.80	52.	15.	0.3
3	5.00	2.01	2.178	0.574	140.	480.	620.	96.	0.57	1.23	101.	59.	0.6
4	5.00	2.01	2.261	0.553	157.	539.	696.	108.	0.71	1.49	90.	45.	0.5
5	5.00	2.01	2.346	0.533	164.	562.	726.	112.	0.85	1.75	81.	32.	0.4
6	5.00	2.01	2.428	0.515	164.	561.	725.	112.	0.98	2.01	73.	21.	0.3
3	5.50	2.01	2.285	0.602	162.	553.	715.	101.	0.59	1.29	116.	68.	0.6
4	5.50	2.01	2.372	0.580	182.	621.	803.	113.	0.74	1.56	104.	52.	0.5
5	5.50	2.01	2.460	0.559	189.	648.	838.	118.	0.89	1.84	93.	37.	0.4
3	6.00	2.01	2.386	0.629	184.	631.	815.	105.	0.62	1.35	132.	78.	0.6
4	6.00	2.01	2.477	0.606	207.	708.	915.	118.	0.78	1.63	118.	59.	0.5
3	6.50	2.01	2.484	0.654	208.	711.	919.	109.	0.65	1.40	149.	88.	0.6
3	3.50	2.50	1.937	0.452	82.	282.	364.	81.	0.45	0.97	57.	37.	0.7
4	3.50	2.50	2.002	0.437	95.	324.	419.	93.	0.56	1.18	52.	29.	0.6
5	3.50	2.50	2.069	0.423	101.	347.	448.	99.	0.67	1.39	47.	23.	0.5
6	3.50	2.50	2.135	0.410	104.	356.	459.	102.	0.78	1.60	43.	17.	0.4
3	4.00	2.50	2.070	0.483	101.	344.	445.	86.	0.48	1.03	70.	45.	0.7
4	4.00	2.50	2.140	0.467	116.	396.	511.	99.	0.60	1.26	63.	36.	0.6
5	4.00	2.50	2.212	0.452	124.	424.	547.	106.	0.72	1.49	57.	28.	0.5
6	4.00	2.50	2.282	0.438	127.	434.	561.	109.	0.84	1.71	52.	20.	0.4
3	5.00	2.50	2.315	0.540	141.	481.	622.	96.	0.53	1.16	97.	63.	0.7
4	5.00	2.50	2.393	0.522	162.	553.	715.	111.	0.67	1.41	88.	50.	0.6
5	5.00	2.50	2.473	0.505	173.	592.	765.	118.	0.80	1.66	80.	39.	0.5
3	5.50	2.50	2.428	0.566	162.	555.	717.	101.	0.56	1.21	112.	73.	0.7
3	3.50	3.00	2.047	0.428	82.	280.	362.	80.	0.42	0.92	55.	38.	0.7
4	3.50	3.00	2.108	0.415	96.	328.	423.	94.	0.53	1.12	50.	32.	0.6
5	3.50	3.00	2.172	0.403	104.	357.	462.	102.	0.64	1.33	46.	25.	0.6
6	3.50	3.00	2.235	0.392	109.	374.	483.	107.	0.75	1.53	42.	20.	0.5
3	4.00	3.00	2.188	0.457	100.	342.	442.	86.	0.45	0.98	67.	47.	0.7
4	4.00	3.00	2.254	0.444	117.	400.	517.	100.	0.57	1.20	61.	39.	0.6
5	4.00	3.00	2.322	0.431	127.	436.	564.	109.	0.69	1.42	56.	31.	0.6
6	4.00	3.00	2.389	0.419	133.	456.	590.	114.	0.80	1.64	52.	24.	0.5
3	5.00	3.00	2.446	0.511	140.	478.	618.	96.	0.50	1.09	94.	65.	0.7

PROPELLANT TANK PARAMETRICS - OPTIMUM DESIGN

TABLE B1

3	3.50	1.50	1.788	0.489	104.	256.	360.	73.	0.56	1.13	56.	30.	0.5
4	3.50	1.50	1.851	0.473	117.	287.	404.	82.	0.67	1.34	50.	22.	0.4
5	3.50	1.50	1.917	0.456	91.	225.	317.	64.	0.78	1.55	45.	0.	0.0
6	3.50	1.50	1.984	0.441	99.	244.	343.	70.	0.88	1.76	41.	0.	0.0
3	4.00	1.50	1.912	0.523	127.	313.	440.	78.	0.60	1.21	68.	37.	0.5
4	4.00	1.50	1.978	0.505	142.	351.	494.	88.	0.71	1.43	61.	27.	0.4
5	4.00	1.50	2.050	0.488	112.	275.	387.	69.	0.83	1.66	55.	0.	0.0
6	4.00	1.50	2.121	0.471	121.	298.	419.	74.	0.94	1.89	50.	0.	0.0
3	5.00	1.50	2.137	0.585	177.	437.	615.	87.	0.68	1.35	95.	51.	0.5
4	5.00	1.50	2.212	0.565	199.	491.	690.	98.	0.80	1.60	86.	37.	0.4
5	5.00	1.50	2.292	0.545	156.	385.	541.	77.	0.93	1.86	77.	0.	0.0
6	5.00	1.50	2.372	0.527	169.	416.	585.	83.	1.05	2.11	69.	0.	0.0
3	5.50	1.50	2.242	0.613	205.	505.	709.	92.	0.71	1.42	109.	59.	0.5
4	5.50	1.50	2.320	0.593	230.	566.	796.	103.	0.84	1.68	99.	43.	0.4
5	5.50	1.50	2.403	0.572	180.	444.	624.	81.	0.97	1.95	89.	0.	0.0
6	5.50	1.50	2.487	0.553	195.	480.	675.	87.	1.11	2.21	80.	0.	0.0
3	6.00	1.50	2.341	0.641	233.	575.	808.	96.	0.74	1.48	125.	67.	0.5
4	6.00	1.50	2.423	0.619	262.	645.	907.	108.	0.88	1.75	112.	49.	0.4
3	6.50	1.50	2.437	0.667	263.	648.	911.	100.	0.77	1.54	141.	76.	0.5
3	3.50	2.01	1.909	0.458	106.	261.	367.	75.	0.53	1.06	54.	33.	0.6
4	3.50	2.01	1.968	0.445	122.	301.	423.	86.	0.63	1.26	49.	26.	0.5
5	3.50	2.01	2.030	0.431	130.	321.	451.	92.	0.73	1.47	45.	19.	0.4
6	3.50	2.01	2.094	0.418	133.	327.	460.	93.	0.84	1.67	41.	14.	0.3
3	4.00	2.01	2.041	0.490	129.	319.	449.	80.	0.57	1.13	66.	41.	0.6
4	4.00	2.01	2.103	0.475	149.	368.	517.	92.	0.67	1.34	60.	32.	0.5
5	4.00	2.01	2.171	0.461	159.	392.	551.	98.	0.78	1.57	55.	24.	0.4
6	4.00	2.01	2.238	0.447	162.	400.	562.	100.	0.89	1.79	50.	17.	0.3
3	5.00	2.01	2.282	0.548	181.	446.	627.	89.	0.63	1.27	92.	57.	0.6
4	5.00	2.01	2.352	0.532	208.	514.	723.	103.	0.75	1.50	84.	45.	0.5
5	5.00	2.01	2.427	0.515	222.	548.	771.	110.	0.88	1.75	76.	33.	0.4
3	5.50	2.01	2.393	0.575	209.	515.	723.	94.	0.66	1.33	106.	65.	0.6
4	5.50	2.01	2.465	0.557	241.	593.	834.	108.	0.79	1.58	97.	51.	0.5
3	6.00	2.01	2.499	0.600	238.	586.	824.	98.	0.69	1.39	121.	75.	0.6
3	3.50	2.50	2.018	0.434	106.	262.	368.	75.	0.50	1.00	52.	35.	0.7
4	3.50	2.50	2.074	0.422	125.	308.	432.	88.	0.60	1.19	48.	29.	0.6
5	3.50	2.50	2.133	0.410	136.	335.	471.	96.	0.70	1.40	44.	23.	0.5
6	3.50	2.50	2.194	0.399	141.	349.	490.	100.	0.80	1.60	41.	17.	0.4
3	4.00	2.50	2.157	0.464	130.	320.	450.	80.	0.54	1.07	64.	43.	0.7
4	4.00	2.50	2.217	0.451	152.	376.	528.	94.	0.64	1.28	59.	35.	0.6
5	4.00	2.50	2.281	0.438	166.	409.	575.	102.	0.75	1.49	54.	28.	0.5
6	4.00	2.50	2.345	0.426	173.	426.	599.	107.	0.85	1.71	50.	21.	0.4
3	5.00	2.50	2.412	0.518	181.	447.	628.	89.	0.60	1.20	89.	60.	0.7
4	5.00	2.50	2.478	0.504	213.	525.	738.	105.	0.71	1.43	82.	49.	0.6
3	3.50	3.00	2.124	0.412	106.	260.	366.	74.	0.48	0.95	51.	36.	0.7
4	3.50	3.00	2.177	0.402	126.	310.	436.	89.	0.57	1.14	47.	30.	0.6
5	3.50	3.00	2.234	0.392	139.	343.	482.	98.	0.67	1.33	44.	25.	0.6
6	3.50	3.00	2.291	0.382	147.	363.	511.	104.	0.76	1.53	40.	20.	0.5
3	4.00	3.00	2.270	0.440	129.	318.	447.	80.	0.51	1.02	62.	44.	0.7
4	4.00	3.00	2.327	0.430	154.	379.	533.	95.	0.61	1.22	58.	37.	0.6
5	4.00	3.00	2.388	0.419	170.	419.	589.	105.	0.71	1.43	53.	31.	0.6
6	4.00	3.00	2.449	0.408	180.	444.	624.	111.	0.82	1.63	49.	25.	0.5

PROPELLANT TANK PARAMETRICS - TARGET DESIGN

TABLE B2

Table B1 considers the 'optimum' tank design using a virtual density of 16.1 lbm/ft^3 and μ = 0.8; Table B2 considers the 'target' design using a virtual density of 21.6 lbm/ft^3 and spherical heads, μ = 1. The program rejected values for which R exceeded 2.5 ft.

Note that in Tables B1 and B2 values for gamma = 1.5, N = 5, 6 do not represent N + N nests.

Values of interest have been compiled from Tables B1 and B2 into a short Table B3. Figure B4 summarizes data from Table B1: the lower four curves are hydrazine masses WP, the upper four combined hydrazine and tank masses W. The radius limit shown is approximate. Shading distinguishes the number of long tanks.

Note that for a given N there is little leeway to change minimum cross section and retain maximum propellant loading. The effect of a wide range of gamma gives about 50 lbm or .25 ft^2 leeway in optimum design. N = 3 or N = 4 seem to be the practical choices.

The general effect of γ on design is seen in Figure B5. The shape of the curves does not depend on A, and so slightly on μ it is not plotted for μ = 1. Design in the flat portions of the curves allows adjustment without major effects.

OPTIMUM DESIGN (TABLE BI)

Wmax = 1200 - 329 = 871

WP	N	A	G	W
667	4	6.	1.5	862
647	5	5.5	2	837
630	3	6.	2	815
621	4	5.5	2	802
616	3	6	1.5	795

TARGET DESIGN (TABLE B2)

Wmax = 1200 - 403 = 797

566	4	5.5	1.5	796
548	5	5.	2.	770
525	4	5.	2.5	738
514	3	5.5	2.	723
514	4	5.	2.	722

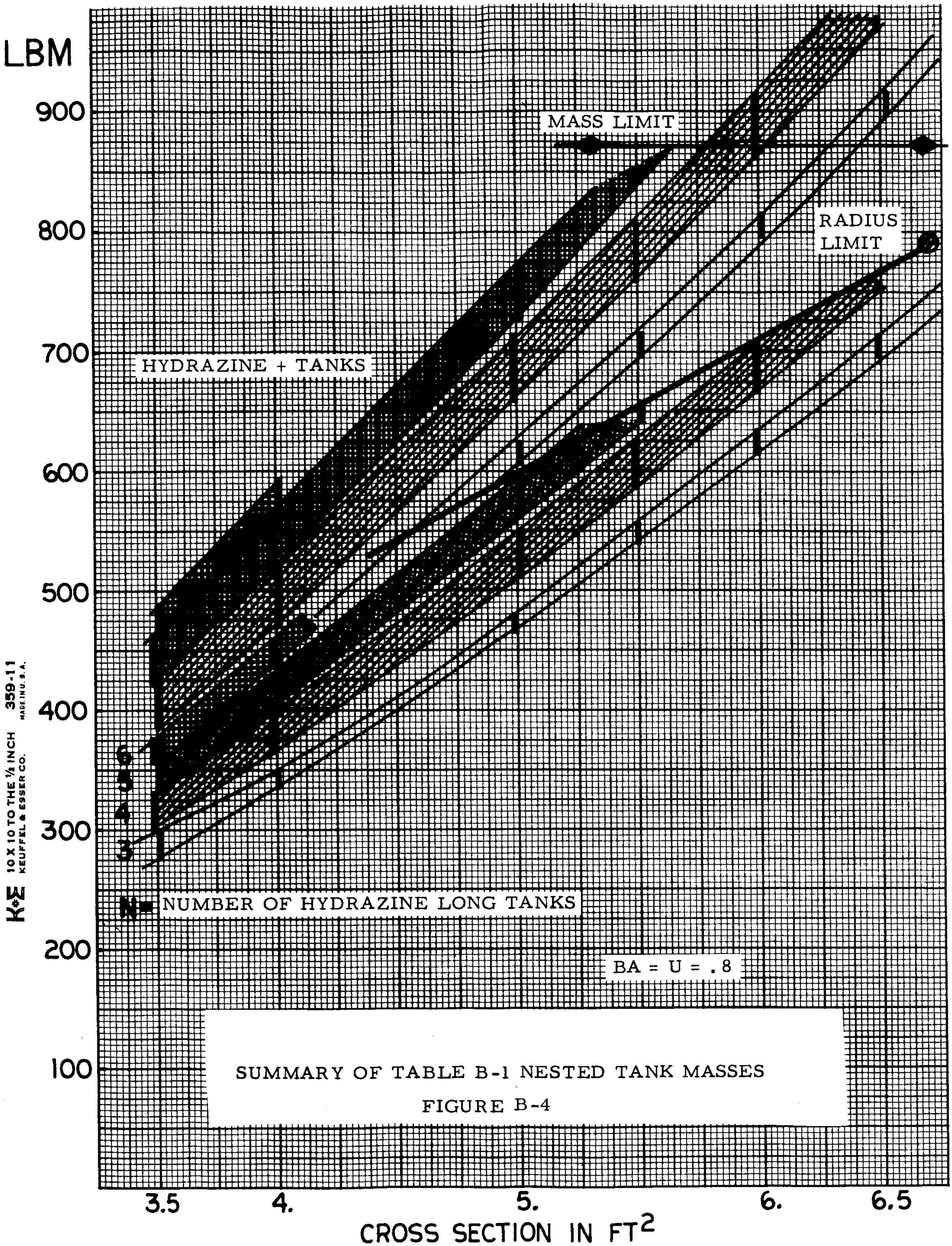
700 DESIGN (TABLE B2)

Wmax = 700 - 289 = 411

287	4	3.5	1.5	404
272	5	4	1.5	387
262	3	3.5	2.5	368
260	3	3.5	3.	366
256	3	3.5	1.5	360

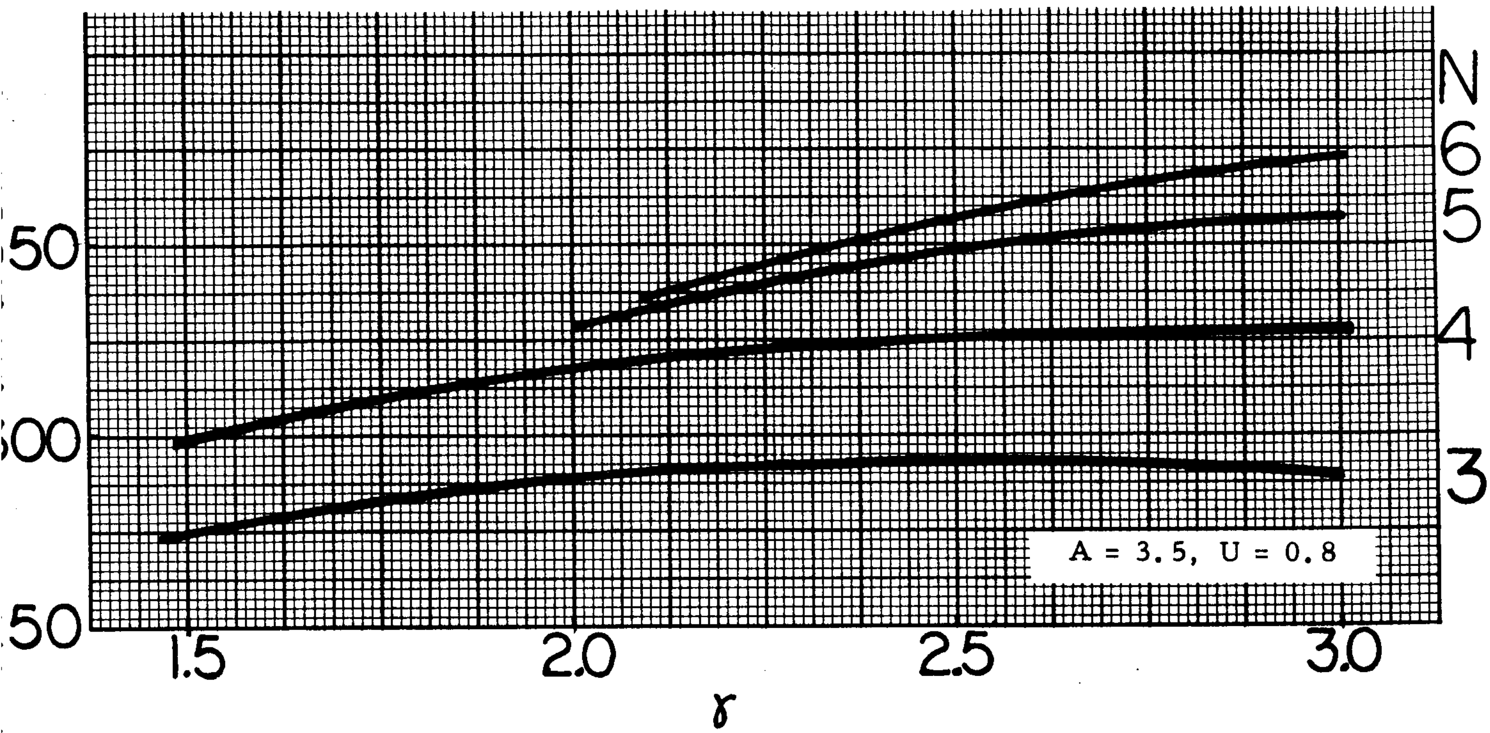
SHORT TABLE OF TANK CONFIGURATIONS

TABLE B-3



SUMMARY OF TABLE B-1 NESTED TANK MASSES

FIGURE B-4



PROPELLANT MASS IN N+N TANKS VS. TANK γ

Shape Representative for all A, U.

FIGURE B-5

APPENDIX C

POWER SOURCE CALCULATIONS AERONOMY/550

The solar cell array consists of 11 percent (air mass 1) N on P cells connected in a series-parallel arrangement to give approximately 28 volts output. The details of solar cell efficiency and loss calculations are given in figure C-1.

The total solar cell array area is 37 square feet (3.44 square meters) yielding approximately 330 watts maximum to the system. The solar cells are mounted on four movable panels 40" X 30" each to provide optimum utilization of incident sun light by orientation of the panels to resolve the effects of regression of orbital nodes as a function of time.

Since the spacecraft is earth oriented, the angle, Θ, between the plane of the solar panels and the sun's rays is constantly changing. The output of the solar cell array is a function of the area A_o projected normal to the sun's rays or $A_o \sin \Theta$. The output power from the array is then $P_o A_o \sin \Theta$, where P_o is the unit power in watts/m^2. The battery is required to be recharged during the illuminated portion of the orbit. The angle associated with P_2 is Θ_2 (see figure c-2). An energy balance equation may be written at this point as:

$$E_f P_o A (\pi + 2\Theta_1) \sin \Theta_1 = 2 \int_{\Theta_1}^{\Theta_2} P_o A \sin \Theta \, d\Theta \qquad (1)$$

$$+ P_o A (\pi - 2\Theta_2) \sin \Theta_2$$

$$- P_o A (\pi - 2\Theta_1) \sin \Theta_1$$

E_f = battery efficiency

The maximum recharge rate of the battery is a function of battery capacity and the state of battery charge. The power matching point (the angle at which the solar panel output power is equal to the average power required

SOLAR POWER AVAILABLE	140 mw/cm^2
Cell Efficiency at AM-1 = 11 percent	
Efficiency Correction at AM-O = 86 percent	
CONVERTED POWER	13.2 mw/cm^2

Losses[1]

Mismatch	8 %	
Temp. at 50 °C (0.48%/°C over 25°C)	12 %	
	20	- 2.6

NET POWER AVAILABLE TO CELLS	10.6 mw/cm^2

Available Packaging Area = 34,400 cm^2
Packaging Efficiency = 90% = 30,960 cm^2

TOTAL POWER FROM PANEL	330 watts

(Solar Incidence Normal to Panel Plane)

[1] Radiation losses at 185 km orbit negligible

SOLAR CELL EFFICIENCY CONSIDERATIONS

FIGURE C-1

by the spacecraft) is denoted as Θ_1. The battery charging characteristics requires that all excess current flow into the battery. The power associated with this current is denoted as P_t and defined from figure c-2 as $(P_2 - P_1)$.
6-12 as $(P_2 - P_1)$.

Equation (1), when integrated and simplified yields:

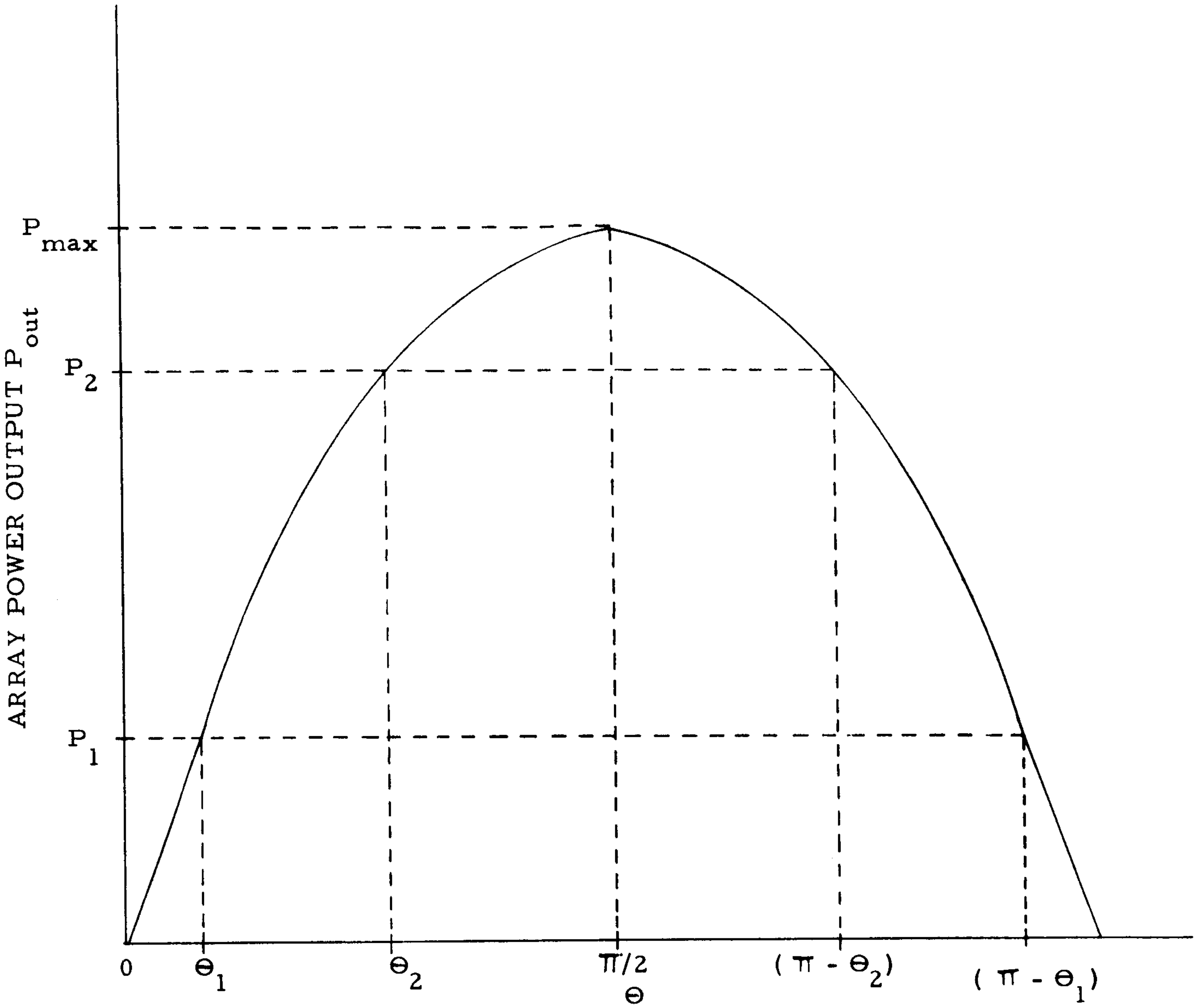
$$
\begin{aligned}
E_f (\pi + 2\Theta_1) \sin \Theta_1 = {} & 2 \cos \Theta_1 - 2 \cos \Theta_2 \\
& + (\pi - 2\Theta_2) \sin \Theta_2 \\
& - (\pi - 2\Theta_1) \sin \Theta_1
\end{aligned}
\tag{2}
$$

Equation (2) provides the basis for the power system definition. The equation was solved for various battery efficiencies and plotted. Another basic equation which relates the maximum charge rate for the battery is:

$$P_o A (\sin \Theta_2 - \sin \Theta_1) = P_t \tag{3}$$

which was included in equation (1) by integrating between the limits of Θ_1 and Θ_2.

The major constraint at this point is the maximum battery charging rate which defines the amount of energy returned to the battery. The state of charge of the battery and its capacity determine the ampere hours that can be useful for charging without creating excessive internal gas pressures. The battery should be cycled (charged and discharged) so that the charged state is less than 75 percent capacity to prevent the gas buildup at high charge rates. A 2OAH capacity battery (50 lbs) was chosen on the basis of the above constraints to be operated from the charged and discharged states of 60 percent and 45 percent, respectively, of the rated ampere-hour capacity. The system will be designed to take a maximum of 10 amps (C/2) while recharging the battery.



$P_1 = P_{av}$ = Average Power available for spacecraft

$P_2 - P_1 = P_t$ Total power differential allowed by battery charging characteristics

P_{max} = Peak power from solar cell array

SOLAR ARRAY CHARACTERISTICS

FIGURE C-2

The C/2 charge allows $\Theta 2$ to be chosen as 90^{o}, which determines Θ_1 as 14.5^{o}. If Θ_1 equals 14.5^{o} the average power available to the spacecraft is $P_o A \sin \Theta_1$ or approximately 66 watts.

1 cy

Space Craft, Inc.

HUNTSVILLE, ALABAMA • CULLMAN, ALABAMA • HOUSTON, TEXAS • GREENVILLE, SOUTH CAROLINA